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Final Report

PRELIMINARY DEFINITION PHASE

APOLLO EXTENSION SYSTEM

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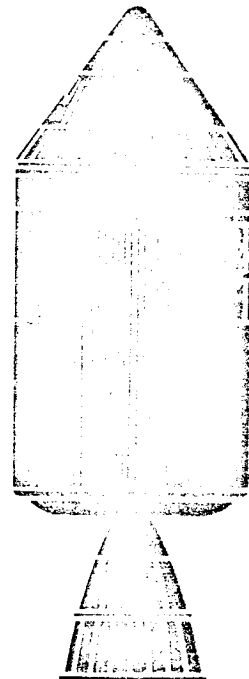
Subsystem
Summary

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Final Report

Preliminary Definition Phase
Apollo Extension System

SUBSYSTEMS SUMMARY

17 January 1966   (✓)

Approved by

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SPACE and INFORMATION SYSTEMS DIVISION

TECHNICAL REPORT INDEX/ABSTRACT

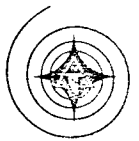| ACCESSION NUMBER						DOCUMENT SECURITY CLASSIFICATION	

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SUBSYSTEMS SUMMARY	

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DESCRIPTIVE TERMS

APOLLO EXTENSION SYSTEM (AES), AES CRYOGENIC STORAGE SYSTEM, AES SM REACTION CONTROL SYSTEM, AES GUIDANCE AND CONTROL SYSTEM, AES COMMUNICATIONS AND DATA SYSTEM, MOLECULAR SIEVE, AES ENVIRONMENTAL CONTROL SYSTEM, TWO GAS ATMOSPHERE, AES DISPLAYS AND CONTROL SYSTEM

ABSTRACT

THIS REPORT SUMMARIZES THE RESULTS OF THE SUBSYSTEM STUDIES PERFORMED DURING THE PRELIMINARY DEFINITION PHASE OF THE AES PROGRAM.

THE SUBSYSTEMS ANALYSES SUMMARIZED IN THIS VOLUME ARE: (1) GUIDANCE AND CONTROL SYSTEM, (2) THE COMMUNICATIONS AND DATA SYSTEM, (3) THE INSTRUMENTATION, AND DISPLAYS AND CONTROL SYSTEMS, (4) ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEMS, (5) CSM THERMAL ANALYSES, (6) THE POWER DISTRIBUTION AND CONTROL, AND THE POWER GENERATION SYSTEMS, (7) THE CRYOGENIC STORAGE SYSTEM, (8) THE SERVICE PROPULSION SYSTEM, AND (9) THE CM AND SM REACTION CONTROL SYSTEMS.

THE STUDIES CONSISTED OF SEVERAL DISTINCT PHASES: (1) DETERMINE THE BLOCK II SUBSYSTEMS CAPABILITIES, (2) ESTABLISH THE AES REQUIREMENTS, (3) DEFINE THE MINIMUM MODIFICATIONS TO THE BLOCK II SUBSYSTEMS TO ACHIEVE THE AES REQUIREMENTS, AND (4) DEFINE THE EXCESS CAPABILITY OF THE SUBSYSTEMS WHICH MIGHT BE USED FOR EXPERIMENTAL PURPOSES.
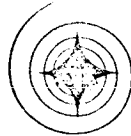
FOREWORD

This document is submitted by the Space and Information Systems Division (S&ID) of North American Aviation, Incorporated, to the National Aeronautics and Space Administration Manned Spacecraft Center in partial fulfillment of the final reporting requirements of Contract NAS9-5017, "Preliminary Definition Study for Utilization of CSM for AES."

Reports being submitted under the subject contract are listed below. Data resulting from subcontractor studies or provided by other sources external to S&ID are included in the appropriate volumes. The reader is urged to refer to other documents in the final report series for further information not contained in this document.

Report No.	Title
SID 65-1145	Master Program Plan (Preliminary)
SID 65-1146	Manufacturing Plan (Preliminary)
SID 65-1147	Facilities Plan (Preliminary)
SID 65-1148	General Test Plan (Preliminary)
SID 65-1150	Configuration Management Plan (Preliminary)
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Report No.	Title
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SID 65-1539	Ground Support and Logistics
SID 65-1541	Interface Methodology
SID 65-1542	Functional Flow Diagrams—Lunar Polar Orbit Reference Mission
SID 65-1543-1	Allis-Chalmers Fuel Cell Study—Technical Summary
SID 65-1543-2	Allis-Chalmers Fuel Cell Study—Program Analysis
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SID 65-1544-2	Land Landing System—Program Analysis
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INTRODUCTION

This volume summarizes the results of the subsystems investigations performed during the Preliminary Definition Phase of the AES program. Detailed descriptions and analyses performed for each subsystem are contained in the applicable reports listed in the Foreword.

The subsystems studies consisted of the following distinct tasks: (1) determining the capabilities of the Block II subsystem configuration, (2) establishing the AES Phase II mission requirements, (3) defining the minimum modifications to the Block II subsystems that would satisfy AES requirements, and (4) defining the excess capability of the subsystems which might be used for experimental functions.

The Block II capabilities were determined by monitoring the progress of the Apollo subsystems. Their status as of September 1965 was used as the baseline from which the AES studies proceeded.

The AES requirements analyses were performed under the constraints that only housekeeping and transit functions were to be considered. Housekeeping functions are those required to sustain the spacecraft in orbit, independent of experimental functions. Transit functions are those required to move the spacecraft into and out of the desired orbit, e.g., those functions required from liftoff to earth-orbit insertion and those required in translunar or transearth phases. Based on the defined housekeeping and transit requirements, excesses or deficiencies in the Block II subsystems were defined in terms of performance, reliability, life, or expendables.

The major Block II subsystem which was found to be deficient in performance for AES was the environmental control system, which is currently not configured to supply a two-gas atmosphere. The Block II subsystems which were judged deficient for AES from a reliability standpoint were guidance and control, communications and data, power distribution and conversion, and environmental control. Based on defined mission success goals, these subsystems were deficient due to the longer operating times or required cyclic operation. The Block II power generation system was found to be life-limited since Block II fuel cells have a life of less than 700 hours. Additional Block II subsystems which were found to be lacking from the standpoint of expendables capacity included the cryogenic storage system, the service module reaction control system, and the environmental control and life support subsystems. The latter subsystems exhibit limited Block II storage space for lithium hydroxide, food, and waste management.

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To obviate these deficiencies, tradeoffs to establish required changes which represented minimum modifications from the Block II baseline were performed. Factors considered in the tradeoffs were reliability, cost, weight, volume, and development schedule. Based on these tradeoffs, appropriate modifications to the Block II subsystems were recommended and integrated into the single selected CSM configuration.

The excess capability that most AES subsystems possess is either inherent in the Block II design or results from the designated AES modifications. For example, the Block II communications and data system has a television and VHF-AM voice capability that is not planned for use in AES missions; this system also has telemetry capability which could be used on an interleaving basis. There is also excess time capability, since AES housekeeping functions use the system for only short periods on a once-per-two-hour basis. The AES guidance and control system possesses certain experimental capability on earth orbital missions in terms of attitude hold or maneuvering, because the specified AES system modifications were based on more stringent lunar orbital mission requirements. The SM RCS contains excess propellant which could be used to satisfy virtually all experimental maneuvers presently defined. Additionally, the power and cryogenic storage systems possess sufficient reactants to supply a LEM laboratory with electrical power and atmosphere on all missions.

GUIDANCE AND CONTROL SYSTEM

GUIDANCE AND CONTROL SYSTEM

The primary objective of the study of the AES guidance and control (G&C) system was to determine the requirement for modifying or adding to the Apollo Block II G&C system in order to accomplish AES housekeeping and transit functions. A secondary objective was to define excess capabilities available for support of experimental activities. The basic guideline for the definition of modifications and excess capabilities was that modifications for product improvement or modifications to enhance excess capability were to be avoided.

The Block II G&C capabilities were compared with the AES requirements, and modifications were defined. The AES requirements are presented in terms of operating timelines and reaction control system propellant consumption.

The G&N performance was also investigated for a worst case mission of a transearth return trajectory from lunar polar orbit. It was assumed that the primary system (guidance and navigation) had failed and that the backup system (stabilization and control) was employed.

For a more detailed description of the G&C study, see NAA S&ID report SID 65-1520.

AES REQUIREMENTS

The four AES reference missions differ significantly from the Apollo LOR mission, which serves as the design reference mission for the Block II systems in both total mission duration and total system operating time.

All four AES missions are at least twice as long as the 8.3-day Apollo LOR mission. The two earth orbital missions impose the most stringent requirement of a 45-day total mission duration. The extended mission duration has a direct effect on the GN&C equipment, since several of its units (i.e., portions of the inertial measurement unit, IMU, power and servo assembly, PSA, and the CM computer, CMC) operate continuously under Block II mechanization throughout the mission. The extended mission duration does not directly affect the SCS equipment, since all units can be turned completely off when not required. However, the extended mission duration does indirectly affect the SCS equipment in that, due to the increased mission duration, the housekeeping functions must be performed for a longer period.

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PERFORMANCE REQUIREMENTS

All of the AES missions have trajectory variations from the Apollo lunar orbit rendezvous mission. These variations occur in various degrees for different AES missions. In order that the impact of these differences on the G&C performance requirements may be assessed, the AES mission phases which require analysis and/or computer program changes have been identified and are shown in Table 1. In each case, the analysis will be designed either to confirm the ability of the G&C to satisfy the appropriate performance requirements or to identify any penalties that are involved. The penalties will take the form of additional fuel requirements, computer program changes, or operational restrictions.

A summary of the analysis requirements is shown in Table 2. It can be seen from this table that most of new mission functions involved are associated with the lunar polar orbit (LPO) mission. The greatest penalties are also expected to be associated with this mission.

G&C OPERATING PROFILES

Analyses to estimate component operating times and SM reaction control system requirements were conducted. The results of the studies are summarized in tabular form for each of the reference missions. (Details concerning the computation of reaction control requirements and the documentation of the individual mission operational sequences are contained in SID 65-1520.)

The total system operating time results from both transit and housekeeping functions. The transit requirements for the two lunar missions are quite similar to the Apollo LOR Block II requirements, while those for the two earth orbit missions are less severe. The housekeeping functions (i.e., passive thermal control, orbit corrections, and attitude hold during solo astronaut sleep periods) are more stringent than those for the Apollo Block II. On the Apollo LOR mission, passive thermal control is provided only during transit. On some AES missions, passive thermal control is required during transit and in orbit between experiments. Orbit corrections are made on both the earth synchronous and lunar polar orbit missions. Attitude hold requirements, to allow communications while the solo astronaut sleeps, are more stringent for the LEM escort mission, since the LEM remains on the moon for almost 14 days instead of less than two days.

From a reliability standpoint, the lunar polar orbit mission is the most severe for the GN&C system, while the LEM escort is the most severe for the SCS. Since both systems are designed for their most severe missions, each system will have excess capability (i.e., operating time capability in excess of transit/housekeeping requirements) for their respective nondesign missions. This excess capability can be used for experiment support.

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Mission Phases and Operating Modes	Apollo
I. Prelaunch	
A. Checkout and alignment	X
II. Ascent to LET jettison	
A. Ascent monitor	X
B. Saturn guidance takeover	X
III. LET jettison to orbit inject	
A. Ascent monitor	X
B. Saturn guidance takeover	X
IV. Service Module abort	
A. Automatic steering	X
B. Manual steering	X
C. SCS delta velocity	X
V. Earth orbit injection with SPS	
A. Automatic steering	-
B. SCS delta velocity	-
C. Manual steering	-
VI. Earth orbit attitude control	
A. S-1VB control	X
B. G&N local vertical	X
C. G&N attitude hold	X
D. SCS attitude hold	X
E. G&N manual attitude control	X
F. SCS manual attitude control	X

Mission Phases and Operating Modes	Apollo
XIII. Midcourse attitude reference alignment	
A. IMU alignment	X
B. GDC alignment	X
XIV. Midcourse correction	
A. G&N/SPS automatic	X
B. SCS/SPS manual	X
C. G&N/RCS automatic	X
D. SCS/RCS manual	X
E. Manual TVC	X
XV. Midcourse attitude control	
A. G&N automatic	X
B. G&N manual	X
C. SCS manual	X
XVI. Midcourse abort (outbound)	
A. G&N delta velocity	X
B. SCS delta velocity	X
C. Manual TVC	X
D. RCS corrections (for free return trajectory only)	X
E. LEM delta velocity	X
XVII. Midcourse plane change	
A. G&N delta velocity	-
B. SCS delta velocity	-
C. Manual TVC	-
XVIII. Lunar orbit injection or EO circularization	
A. G&N delta velocity	X
B. SCS delta velocity	-

Table 1. G&C Mission Functions for Apollo and AES

| | | Missions | | | | | |
Mission Phases and Operating Modes		Apollo	ELI	EPO	ESO	LPO	LLE
VII.	Earth orbit attitude reference alignment						
	A. IMU coarse and fine alignment	X	X	X	X	X	X
	B. BMAG/GDC alignment	X	X	X	X	X	X
VIII.	Earth orbit navigation						
	A. MSFN data	X	X	N	N	N	X
	B. Landmark sightings	X	X	N	N	N	X
	C. Star horizon sightings	X	X	N	N	N	X
	D. Stored data	X	X	C	C	C	X
IX.	Deboost from earth orbit						
	A. G&N delta velocity	X	X	X	N	X	X
	B. SCS delta velocity	X	X	X	C	X	X
	C. Manual TVC	X	X	X	C	X	X
	D. RCS delta velocity	X	X	C	C	X	X
X.	Translunar injection or orbital transfer						
	A. G&N monitor	X	–	N	N	N	X
	B. Saturn guidance takeover	X	–	C	C	C	X
	C. G&N abort	X	–	C	C	C	X
	D. SCS abort	X	–	C	C	C	X
XI.	Transposition and docking						
	A. G&N manual control	X	–	–	–	N	N
XII.	Midcourse navigation						
	A. MSFN data	X	–	–	N	X	X
	B. Star landmark sightings	X	–	–	C	C	X
	C. Star horizon sightings	X	–	–	C	C	X

| | | Missions | | | | | |
Mission Phases and Operating Modes		Apollo	ELI	EPO	ESO	LPO	LLE
XIX.	Lunar orbit attitude control						
	A. G&N local vertical	X	–	–	–	X	X
	B. G&N attitude control	X	–	–	–	X	X
	C. SCS attitude control	X	–	–	–	X	X
	D. Manual attitude control	X	–	–	–	X	X
XX.	Lunar orbit attitude reference alignments						
	A. IMU alignment	X	–	–	–	X	X
	B. GDC alignment	X	–	–	–	X	X
	C. LEM reference alignment	X	–	–	–	X	X
XXI.	Lunar orbit navigation						
	A. MSFN data	X	–	–	–	N	X
	B. Landmark tracking	X	–	–	–	C	X
	C. Star horizon tracking	X	–	–	–	C	X
XXII.	Backup rendezvous						
	A. G&N delta velocity	X	–	–	–	–	C
	B. SCS delta velocity	X	–	–	–	–	C
	C. Manual TVC	X	–	–	–	–	C
XXIII.	Transearth injection						
	A. G&N delta velocity	X	–	–	–	N	N
	B. SCS delta velocity	X	–	–	–	C	C
	C. Manual TVC	X	–	–	–	C	C
	D. LEM delta velocity	X	–	–	–	–	C
XXIV.	Entry						
	A. G&N entry	X	X	X	X	X	X
	B. EMI	X	X	X	X	X	X
	C. SCS entry	X	X	X	–	–	–
	D. Manual steering	X	X	X	–	–	–

SYMBOLS

X = Existing capability
N = Normal mission function requiring analysis and/or CMC program changes
C = Contingency mission function requiring analysis and/or CMC program changes
– = Not applicable

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Table 2. Analysis and/or Computer Program Changes Required

Normal Mission Functions

    1. Ascent monitor for EPO, ESO, and LPO

    2. EO injection for EPO and ESO

    3. EO navigation for EPO, ESO, and LPO

    4. Translunar injection for LPO

    5. Midcourse corrections for LLE and LPO

    6. LO injection for LLE and LPO

    7. LO navigation for LPO

    8. Transearth injection for LLE and LPO

    9. Midcourse plane change for LPO

  10. Deboost from EO for EPO and ESO.

Contingency Mission Functions

    1. Saturn guidance takeover for EPO, ESO, and LPO

    2. SM abort for EPO, ESO, and LPO

    3. Midcourse abort for ESO, LLE and LPO

    4. Backup rendezvous for LLE.

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Earth Polar Orbit - Reference Mission 1

The orbital phase of this mission will be flown with the longitudinal axis of the vehicle oriented normal to the orbit plane. The vehicle is rolled at orbital rate such that one of the transverse body axes remains continuously oriented along the local vertical. Since this orientation is satisfactory for communication using the omni antennas, there are no additional attitude constraints associated with communication requirements. Thermal control considerations dictate that the vehicle be yawed 180 degrees every two orbits.

All sensing equipment can be located on the underside of the vehicle on the axis which continually faces the earth. The solar panels will be attached to the sides of the vehicle. It has been assumed that they will be so located that aerodynamic disturbance effects will be negligible. The nominal attitude dead band will be ±5 degrees for panel pointing. When an experiment is scheduled, the vehicle will be maneuvered to the required dead band at a rate of 0.02 degree per second, permitting the vehicle to slew from 5 to 0.1 degrees in less than five minutes. If this maneuver were not performed, convergence to the attitude limits necessary for experimentation at the rates encountered during limit-cycling would be excessive. It will not be necessary to perform another maneuver at the termination of the experiment, since the vehicle is already within the ±5-degree nominal attitude for panel orientation.

Earth Synchronous Orbit - Reference Mission 2

Thermal problems on this mission will require the use of a passive thermal control mode at those times when experiment attitude holds are not being performed. If the experiment frequency requirements are examined independently, a maximum of 182 thermal cycling periods are required. However, by performing some experiments sequentially, the cumulative hold time is no greater than one hour (tolerable from a thermal standpoint), and the number of thermal cycling periods can be reduced to 139. It is assumed that the experiments would be so scheduled.

Another housekeeping requirement concerns orbit correction capability. The SM RCS should be capable of providing 40-second burn time from each of two jets.

Communication requirements can be satisfied if additional constraints are imposed on the thermal cycling mode. The basic thermal constraint is satisfied by orienting the longitudinal axis of the vehicle normal to the sun line and rolling at a rate of approximately 0.2 degree per second. The additional communication constraint consists of orienting the longitudinal axis normal to the plane of the ecliptic, thereby ensuring continuous coverage with the S-band omni antenna. The communication constraint does not pose any additional requirements in terms of component operating time or RCS requirements.

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Lunar Polar Orbit - Reference Mission 3

During the lunar orbit phase, thermal considerations dictate the use of passive thermal control during the long duration nonmapping periods. This constraint results in a requirement for 112 thermal cycling periods with the X axis of the vehicle oriented normal to the sun line and the vehicle rolling at 0.25 degree per second. Communications analyses indicate that the proposed thermal cycling will not present a problem with respect to orientation of the high-gain antenna. Furthermore, it is assumed that the mapping orientation requirement takes priority over any communication orientation requirement, and so communication maneuvers have not been scheduled during mapping periods. This should not be interpreted as meaning communication can occur only if the orientation is required for mapping.

Orbit corrections represent another housekeeping function. There are two corrections, each of which involves a firing of the SPS engine. Reference system alignments and navigation sightings are also associated with the corrections.

The experimental program of Mission 3 concerns mapping of the lunar surface. The mapping sequence consists of six hours of mapping in six segments every 24 hours. Since two of the segments follow prior mapping periods by only 15 minutes, it is not necessary to reorient the vehicle. Therefore, the mapping consists of four operational segments each day for 28 days. It is assumed that, prior to each mapping segment, it is necessary to align the vehicle reference system and to align the vehicle to the local vertical.

LEM Escort - Reference Mission 4

The housekeeping functions on this mission are the same as those scheduled for the LOR mission. However, the extended lunar orbit duration results in an increased requirement for inertial stabilization during the solo astronaut sleep periods.

It is assumed that, during the nonsleep periods, the astronaut will provide manual gross vehicle control in order to satisfy communication and ECS radiator orientation constraints. The only units necessary to provide this control are the jet drivers and a rotational hand controller. Assuming intermittent operation, it is estimated that the units will be operated for a cumulative duration of five hours.

Operational Requirements Summary

Tables 3, 4, 5, and 6 contain summaries of the component operating times and the nominal reaction control requirements for each of the reference missions. It is important to note that the reaction control requirements do

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Table 3. Operational Requirements Summary - Reference Mission 1

Component	Transit	House-keeping	Subtotal	Experiment Support	Total
SCS (Operating Time, Hr)					
H/C and RCS driver assembly	4:27	36:00	40:27	1:45	42:12
Gyro assembly No. 1	3:20	-	3:20	1:45	5:05
Gyro assembly No. 2	2:30	-	2:30	-	2:30
Servo amplifier assembly	1:02	-	1:02	-	1:02
Display electronics	4:52	-	4:52	1:45	6:37
Control electronics	2:34	-	2:34	1:45	4:19
GDC assembly	2:30	-	2:30	1:45	4:15
GN&C (Operating Time, Hr)					
IMU	4:52	-	4:52	-	4:52
CMC and DSKYs	4:52	0:24	5:16	-	5:16
Optics	1:36	-	1:36	-	1:36
Attitude impulse	1:33	-	1:33	-	1:33
SM RCS					
Propellant Quantity (Lb)	170.15	1080.70	1250.85	378.91	1629.76
Number of Jet Starts					
Pitch	141.0	-	141.0	3405.0	3546.0
Yaw	27.0	6120.0	6147.0	4062.0	10209.0
Roll	52.0	180.0	232.0	2958.0	3190.0
Jet Burn Time (Sec)					
Pitch	76.61	-	76.61	54.57	131.18
Yaw	64.97	755.0	819.97	65.17	885.14
Roll	4.19	4.40	8.59	42.53	51.12

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Table 4. Operational Requirements Summary - Reference Mission 2

Component	Transit	House-keeping	Subtotal	Experiment Support	Total
SCS (Operating Time, Hr)					
H/C and RCS driver assembly	18:46	35:45	54:31	44:00	98:31
Gyro assembly No. 1	6:54	0:20	7:14	44:00	51:14
Gyro assembly No. 2	6:35	0:20	6:55	-	6:55
Servo amplifier assembly	1:42	0:12	1:42	-	1:42
Display electronics	18:11	1:00	19:11	44:00	63:11
Control electronics	12:47	35:45	48:32	44:00	92:32
GDC assembly	6:35	0:20	6:55	44:00	50:55
GN&C (Operating Time, Hr)					
IMU	18:11	1:00	19:11	-	19:11
CMC and DSKYs	20:41	1:24	22:05	-	22:05
Optics	5:46	0:40	6:26	-	6:26
Attitude impulse	5:30	0:40	6:10	-	6:10
SM RCS					
Propellant Quantity (Lb)	214.96	660.46	875.42	755.82	1631.24
Number of Jet Starts					
Pitch	401.0	2519.0	2920.0	3552.0	6472.0
Yaw	69.0	2519.0	2588.0	3552.0	6140.0
Roll	185.0	1255.0	1440.0	1923.0	3363.0
Jet Burn Time (Sec)					
Pitch	90.73	156.50	247.23	199.90	447.13
Yaw	67.18	156.50	223.68	199.90	423.58
Roll	9.90	83.28	93.18	66.02	159.20

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Table 5. Operational Requirements Summary - Reference Mission 3

Component	Transit	House-keeping	Subtotal	Experiment Support	Total
SCS (Operating Time, Hr)					
H/C and RCS driver assembly	44:02	34:37	78:39	28:00	106:39
Gyro assembly No. 1	8:30	1:22	9:52	28:00	37:52
Gyro assembly No. 2	7:59	1:22	9:21		9:21
Servo amplifier assembly	2:37	0:23	3:00		3:00
Display electronics	17:18	6:37	23:55	28:00	51:55
Control electronics	36:45	29:22	66:07	28:00	94:07
GDC assembly	7:59	1:22	9:21	28:00	37:21
GN&C (Operating Time, Hr)					
IMU	17:18	6:37	23:55		23:55
CMC and DSKYs	19:28	7:01	26:29		26:29
Optics	7:03	2:59	10:02		10:02
Attitude impulse	5:56	2:04	8:00		8:00
SM RCS					
Propellant Quantity (Lb)	252.27	406.78	659.05	1022.38	1681.43
Number of Jet Starts					
Pitch	478.0	2642.0	3120.0	4607.0	7727.0
Yaw	78.0	2332.0	2410.0	4620.0	7030.0
Roll	181.0	1171.0	1352.0	1240.0	2592.0
Jet Burn Time (Sec)					
Pitch	96.90	204.21	301.11	331.73	632.84
Yaw	77.18	9.04	86.22	314.21	400.43
Roll	6.10	34.86	40.96	23.85	64.81

Table 6. Operational Requirements Summary - Reference Mission 4

Component	Transit	House-keeping	Subtotal	Experiment Support	Total
SCS (Operating Time, Hr)					
H/C and RCS driver assembly	56:58	109:42	171:40	-	171:40
Gyro assembly No. 1	19:14	104:42	123:56	-	123:56
Gyro assembly No. 2	8:00	-	8:00	-	8:00
Servo amplifier assembly	2:37	-	2:37	-	2:37
Display electronics	28:15	-	28:15	-	28:15
Control electronics	47:30	104:42	152:12	-	152:12
GDC assembly	8:00	-	8:00	-	8:00
GN&C (Operating Time, Hr)					
IMU	28:15	-	28:15	-	28:15
CMC and DSKYs	27:50	0:24	28:14	-	28:14
Optics	12:47	-	12:47	-	12:47
Attitude impulse	7:51	-	7:51	-	7:51
SM RCS					
Propellant Quantity (Lb)	283.82	343.14	626.96	-	626.96
Number of Jet Starts					
Pitch	465.0	1376.0	1841.0	-	1841.0
Yaw	78.0	1057.0	1135.0	-	1135.0
Roll	152.0	635.0	787.0	-	787.0
Jet Burn Time (Sec)					
Pitch	108.58	94.16	202.74	-	202.74
Yaw	80.34	71.32	151.66	-	151.66
Roll	5.93	35.23	41.16	-	41.16

not include contingencies and that the jet start and jet burn time numbers correspond to the worst case in each control axis. It should also be noted that the component operating times for experiment support include only maneuvering functions. The RCS requirements include both maneuvering and attitude hold functions.

AES G&C CONFIGURATION

The G&C configuration requirements studies for extended operation received more emphasis than any of the other G&C analyses. It was emphasized by NASA at the outset that the SCS and GN&C subsystems were to be used only for transit and housekeeping functions and not for experiment stabilization in orbit. Although it might be supposed that the G&C operating times for housekeeping and transit functions on AES missions will not be markedly different from Apollo, such is not the case. On AES missions lasting up to 45 days, the GN&C subsystem must operate continuously in a standby mode as compared to 8.3 days for Apollo. The SCS must provide all of the housekeeping attitude changes required during the longer periods of orbital operation.

BLOCK II G&C RELIABILITY ON AES MISSIONS

Table 7 shows the reliability of the Block II SCS and GN&C subsystems on the LPO and LLE missions in comparison with the apportioned reliability objectives. The LPO is considered to be the worst case mission for the GN&C and the LLE is considered to be the worst case for the SCS (if it is assumed that the SCS must provide attitude hold during solo astronaut sleep periods). The GN&C is seen to be far below its apportionment goal. It will be shown later that this is primarily due to the low reliability of the standby mode on long duration missions.

GN&C RELIABILITY IMPROVEMENT

The lunar polar orbit mission was selected for detailed GN&C reliability studies because of the high operating times in both the operate and standby modes. Earth orbital missions operate about 30-percent longer in the standby mode, but require less than half the time needed on lunar missions in the operating modes. The use of the LPO mission for apportionment and the greater reliance on the GN&C on lunar missions for crew safety were also factors that entered into the selection of the LPO mission for GN&C reliability analysis.

GN&C reliability improvement was approached in three steps: (1) reduction of GN&C equipment operating time, (2) standby mode improvement, and (3) addition of redundant components. Reduction of operating time was examined first, because, if it could be achieved, the hardware penalties would be least. Standby mode improvements were looked at next, because the standby mode is the greatest source of G&C unreliability on AES missions.

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Table 7.   Predicted Reliability of Block II G&C Subsystems
on AES Missions

| Subsystem | LPO Mission | | LLE Mission |
	Apportionment Goal	Predicted Reliability	Predicted Reliability
Block II GN&C	0.9973	0.909	0.942
Block II SCS	0.9956	0.997	0.973

Reduction of GN&C Equipment Operating Time

Efforts to reduce GN&C equipment operating time in Block II had already progressed to a point of diminishing returns at the time these studies were started in that the Apollo program had attempted to minimize GN&C operating time.   Nevertheless, two additional possibilities were explored to see if significant reliability gains could be achieved by relieving the GN&C of some of the responsibilities it has on the Apollo mission.   First, responsibility for navigation in lunar orbit was transferred to MSFN.   Second, responsibility for performing midcourse corrections was assigned to the SCS. It was expected that a degradation in performance (which would take the form of an increase in SPS propellant) would result from these changes.   The result was to improve the probability of mission success on the LPO mission from 0.909 to 0.920.   This improvement was not considered sufficient to represent an adequate solution, so other means were then examined for achieving reliability gains.

Improvement of CMC Standby Mode Reliability

The command module computer (CMC) standby mode of operation contributes over 75 percent of the total probability of GN&C failure when the Block II system is used on an LPO mission.   The inertial measurement unit (IMU) standby mode is the next largest contributor (11 percent of the total) to unreliability.

The CMC standby mode, also referred to as the CMC clock, is the portion of the CMC which, under Apollo operating rules and mechanization, must operate at all times throughout the mission.   This requirement is necessary so that an absolute time reference can be maintained with sufficient accuracy to assure successful navigation.

On the LPO mission, the CMC standby mode equipment operates 746 hours (including the time it provides support to the CMC normal operation). This operating time can be reduced to 17.5 hours if some means can be found for reestablishing accurate on-board absolute time information after it has been lost. On the LPO mission, this would increase the reliability of a Block II GN&C from 0.909 to 0.975.

A procedure was determined that will permit reestablishment of absolute time information in the CMC after it has been lost. In this procedure, the CMC is restored to operate mode and a time at which the clock in the computer is to be started is determined. A crew member selects the appropriate computer programs to perform the clock restart function, and, upon indication that it is ready, the restart time is entered through the keyboard. At a time that takes into account the transmission and data processing delays, a marker signal is initiated at ground control. This signal is transmitted to the spacecraft over the up-data link (UDL). Upon receipt of the properly coded signal by the computer, the accumulation of time will be restarted automatically. The time, as generated in the computer, will then be sent to ground control over the telemetry link for verification. The accuracy of the clock should be higher, after stopping and restarting, with this procedure than it would be near the end of an Apollo mission. For the longer AES missions, considerably greater accuracy can be achieved by using this procedure than by relying on the on-board frequency standard.

In addition to the UDL, the voice link can also be used for restarting the clock. In the event that a reduced capability occurs in the voice or UDL paths at any time during the mission, the CMC clock will be started and will remain operating for the remainder of the mission. It is recommended that future studies evaluate the feasibility of modifying the central timing equipment to provide an on-board means of restarting the computer clock.

A disadvantage of stopping the computer clock is that it places a stronger dependence upon the MSFN and the communications system to accomplish the navigation function. Stopping the clock results in making portions of the communications system essential in both the primary and backup navigation modes. The primary (MSFN) navigation function utilizes the communication links to provide data to the spacecraft and the backup (GN&C) navigation function utilizes the communication links to update the computer clock. Because of the existing redundancies in the communication links, there are only a few components of the communications subsystem that are necessary for both functions. A failure of any one of these components would result in the loss of the spacecraft and crew on a lunar mission. It would therefore be desirable to provide redundancies for the components involved.

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A modification of the CMC power switching is also recommended. At the present time, power switching for the computer is provided only to select standby or normal operation. An additional power switch will be required to turn the computer completely off. The existing circuit breakers are adequate because they are provided for both normal and standby operation.

Improvement of IMU Standby Mode Reliability

For Block II IMU standby operation, the 3200-cps power supply module in the PSA and the heaters and temperature control circuits in the IMU operate continuously. The functions of the IMU standby mode are to provide temperature control and electromagnetic suspension power for the PIPA's and IRIG's in the IMU. Studies have shown that the IMU standby mode can be modified so that it only provides heater operation and not suspension operation. The resulting reduction in failure rate will be from 14.1 failures per million hours to 7.8 failures per million hours. The temperature control equipment cannot be shut down at any time during the mission since provision will not be available to recalibrate the sensors during flight.

The electromagnetic suspension system (ducosyn) provides both axial and radial centering forces for geometrical stabilization of the PIPA and IRIG floats. Excitation power (3200 cps) is provided from two PSA modules through the IMU slip rings directly to the six sensors. This suspension power is required only during IMU operation, and provision should be made to eliminate it from standby operation on the AES configuration because of the long periods of standby involved. Suspension system operation can be accomplished by wiring changes external to the IMU and would be provided only when full power is applied to the IMU.

The 3200-cps suspension power is synchronized by a reference signal from the CMC. Operation of the 3200 cps power supply is possible without this synchronization signal because it has a free running mode at a lower frequency. The free running frequency, however, approaches the resonant frequency of the PIPA control loops and can cause their operation to become unstable. It will, therefore, be an operational requirement to return the CMC to standby operation prior to applying power to the IMU for normal operation. No physical damage will result from turning off the suspension power.

GN&C Redundancies

Two factors were considered in establishing the order of adding redundancy: (1) the expected reliability gain (measured by the product of failure rate and operating time), and (2) the expected penalty. The penalties associated with the addition of a complete CMC or IMU were judged to be prohibitive and these redundancies were not considered, except as a last

resort. Modification of the IMU will incur development and qualification costs requiring detailed subcontractor studies. For this reason, IMU heater and slip ring redundancy could not be completely appraised even though it appeared to be second in its potential for reliability improvement. It is anticipated that these studies will be completed during future phases. The IMU standby redundancy within the PSA would also incur physical design changes and requalification.

When the CMC clock is not continuously operated, the provision of redundant CDU's and modification of the PSA to provide redundant IMU suspension power will improve the GN&C reliability from 0.975 to 0.983. This configuration (designated configuration A in Table 8) was thought to be the optimum baseline for reliability improvement that reasonably can be attained by the GN&C on the LPO mission. It was this judgment that resulted in reestablishment of the GN&C apportionment (predicted feasible goal) to 0.9833. The difficulty of providing redundant heaters and slip rings in the IMU are not specifically known at present. It appears likely that, with further study, this additional change could be considered feasible. The provision of a redundant CMC would probably require costly redesign of secondary structure in the lower equipment bay and does not appear to be desirable at this time.

An analysis was also made of the CMC clock operating continuously. To meet the original apportionment goal, it would be necessary to add a complete computer, redundant ISS portion of the PSA, CDU's, a redundant CMC clock and to modify the IMU. To meet the new apportionment goal, it would be necessary to design and qualify a new component (redundant CMC clock) and make the IMU modification.

Table 8 shows the weight, volume, and installation tradeoffs associated with some of these redundant configurations. Configuration D in Table 8 has redundant CDU's, and redundant heaters and slip rings in the IMU. The system would be operated with the CMC clock and IMU suspension power turned off when the CMC and IMU, respectively, are not in use. If the IMU modifications do not prove to be costly in dollars or schedule risk, this configuration would provide maximum reliability gain for minimum weight and volume increase.

It should be noted that all redundant configurations will require equipment, in addition to that which has been discussed, to provide the switchover capability from the failed component to its replacement.

RECOMMENDED GN&C CONFIGURATION

Various means were studied for improving the GN&C reliability to compensate for the additional operating time on AES missions. A baseline configuration was established as a result of studies completed prior to

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Table 8. GN&C Configuration Trade-offs

Configuration	A	B	C	D
Redundancies	IMU Standby Elec. (PSA), CDU's No CMC Standby Operation	Also CMC	Also PSA (ISS), IMU Htr. and slip rings	CDU's, IMU Htr. and Slip rings, No CMC standby Operation, Turn Off IMU Standby Elec. (PSA)
LPO Mission Reliability	0.983	0.989	0.998	0.989
Added Volume (Cubic Feet)	0.77	1.81	2.32	0.74
Added Weight (pounds)	35	93	120	33
CM Installation	Minor Change	Significant Relocation of Equipment	Major Redesigns	Minor Change

September 1965. An improved configuration, however, has been developed since the adoption of the baseline configuration. This configuration provides for less additional equipment and greater reliability improvement. All of the additional improvement is obtained in the equipment associated with the ISS standby operation which is accomplished in two ways: first, the PSA standby electronics (IMU suspension power) is turned off completely, leaving only the heater circuitry in the IMU operating during standby; second, the heater circuitry in the IMU is considered to be completely redundant, including heaters, sensors, control circuitry, and slip rings. It is not presently known if complete IMU heater circuitry redundancy is feasible.

The GN&C reliability for the LPO mission will vary between 0.983 and 0.989 depending on the extent of IMU improvement utilized. As an interim configuration, it is recommended that the configuration described above, excluding IMU improvement, be adopted for future studies pending completion of the evaluation of IMU modifications.

The recommended configuration consists of the following: (1) CMC clock shutdown when normal operation is not required, (2) IMU power switching modification to eliminate suspension power during ISS standby, (3) addition of a CMC power switch for turning the computer completely off, (4) redundant CDU package, and (5) redundant CDU switching assembly.

This configuration has a reliability of 0.983 and requires an addition of 0.74 cubic feet, and 33 pounds over Block II.

SCS REDUNDANCY REQUIREMENTS

The baseline SCS configuration for AES missions was defined at the study midpoint to consist of a Block II SCS with a redundant gyro assembly and a redundant solenoid driver assembly. This configuration was established by prior reliability analyses and NASA review of the results. Two key assumptions had been made in the reliability analysis which led to the choice of this configuration. One assumption was that the SCS must stabilize the spacecraft on the LLE mission while the single astronaut sleeps; the other assumption was that the failure of a single jet driver within the solenoid driver assembly would require an abort. The validity of the former assumption may now be questioned because of recent interest by NASA in developing a new system which could provide stabilization during astronaut sleep periods. Because the new system is only in the conceptual stage and time did not permit consideration of it for this role, the reliability analysis was based on the assumption that stabilization during sleep periods would be provided by the SCS. The validity of the second assumption has been a subject of SCS study since September.

Prior to the program midterm, it was assumed that mission success would not be achieved with a jet driver failure. If, however, it is assumed that mission success can be achieved with a driver failure (occurring at any point in the mission) the SCS reliability can be significantly increased without the addition of a redundant driver package. As a result, a new ground rule was established for use in the SCS reliability analysis. This ground rule provided for the continuation of the mission following a jet driver failure.

The validity of the ground rule was judged on the basis of a capability to satisfy the following three criteria:

1.   All mission success functions can be performed adequately with a single driver failure.

2.   A second driver failure will not prohibit achievement of crew safety.

3.   Contingency RCS propellant requirements resulting from a single driver failure are not prohibitive for mission success.

These criteria were then used to deduce vehicle control requirements associated with mission success, crew safety, and contingency propellant requirements. An effort was then made to define a method to supply electrical power to the jet solenoids and their associated drivers in such a manner that the vehicle control requirements were satisfied. The study results indicated that each criterion was satisfied, and the ground rule is considered valid.

All reliability apportionment requirements are based on the LPO reference mission, however, due to the SCS attitude hold requirements imposed by the LLE mission, selection of redundancies was based on this reference mission. The SCS redundancies for this mission were then selected based upon relative failure rates and required operating times. Mission success probabilities were used to establish redundancy requirements since, for the SCS, mission success imposes more stringent reliability requirements than does crew safety. The analysis is based on the best available SCS Block II failure rate predictions made early in 1965. The reliability values used are subject to change upon receipt of new failure-rate data expected to be available early in 1966.

Table 9 contains the mission success probabilities as a function of both reference mission and SCS redundancy, where the redundancy is indicated as standby or operational. For the standby case, it is assumed that power will be applied to the redundant package after a failure of the primary package. For the operational case, it is assumed that power will be applied to the redundant package and the primary package during the same time intervals.

Table 9. SCS Mission Success Reliability

Considered Redundancy	Type		Reliability			
			LLE Mission	LPO Mission		
	Standby	Operational		Transit & Hskg.	Min. Mav.	Max. Mav.
Basic Block II	-	-	0.9733	*0.9970	*0.9922	*0.9874
Gyro	-	X	0.9951	-	*0.9961	*0.9939
Gyro + CEA	X	X	0.9974	-	*0.9964	*0.9942
Gyro + Driver (AES Baseline)	-	X	0.9961	0.9987	*0.9967	*0.9946
Gyro	X	X	0.9956	-	-	-

* Assumes GDC and displays used to perform "Experiment Orientation" maneuvers.

For the redundant gyro assembly configuration, the reliability number was calculated for both standby and operational redundancy, in the case of LLE mission, where the effects may be considered significant due to the long gyro operating times.

The lunar polar orbit mission is subdivided into transit and housekeeping only, minimum experiment maneuvers, and maximum maneuvers. Transit and housekeeping only does not include any operating time for vehicle orientations to support experiments. The minimum and maximum maneuver portions of the mission include transit, housekeeping and those maneuvers required to support one complete mapping and two complete mappings, respectively. The displays, GDC and BMAG's, were used to perform the maneuvers, thus providing a rather precise method of maneuvering which is not considered to be a requirement but is included here to provide an indication of the degradation in reliability associated with the more precise method of maneuvering.

Table 9 shows that the basic Block II configuration on the LLE mission has a reliability of only 0.9733 compared with the preliminary apportionment of 0.9956. This decrease is due primarily to the use of the attitude hold mode while the solo astronaut is asleep. Addition of a redundant gyro assembly brings the reliability to a level sufficiently near the goal. It is preferable that the redundant gyro assembly not be turned on until a gyro failure occurs (from a reliability point of view).

On the LPO mission, a basic Block II configuration exceeds the reliability apportionment when its use is confined to transit and housekeeping. However, when used to supply attitude maneuvers for one cycle of mapping, the reliability decreases to 0.9922. The reliability is only 0.9874 for two cycles of mapping. (The basic mission plan calls for two cycles of mapping.) None of the configurations studied were able to meet the reliability apportionment when required to support two cycles of mapping. The baseline configuration (redundant gyro assembly and solenoid driver assembly), with a reliability of 0.9946, most nearly met the reliability apportionments.

The reliability apportionments for the LPO mission presume the use of the gyro display coupler and the SCS displays for attitude change maneuvers. If these maneuvers can be made with the rotation control and jet drivers, without the use of displays, it is expected that the Block II configuration can satisfy the apportionment for two cycles of mapping.

None of the reliability values previously indicated consider the degradation that will result when the necessary switching is added to effect the switchover between packages. This degradation could be quite small, if, for example, the switching is manual and is located at the boxes in the lower equipment bay.

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The SCS redundancy requirements are based on the mission success probability goals for the LLE mission. These redundancy requirements are imposed by the long-term attitude hold intervals during the lunar orbit astronaut sleep periods. The reliability goal was not used as a hard-and-fast requirement but rather as a guide in determining the recommended level of redundancy.

The selection of the redundancy level (basic Block II plus redundant gyro assembly) is based on the trade-off data shown in Table 10.

Considering reliability versus design impact, it is noted that the probability of the occurrence of an SCS failure (which would cause a mission abort) is 26.7 times per 1000 missions. The addition of the gyro assembly reduces this rate to 4.4 per 1000 missions; the addition of the gyro and solenoid driver assemblies reduces the rate to 3.9 per 1000 missions; the addition of the gyro and CEA assemblies reduces the rate to 2.6 per 1000 missions. The point, then, of rapidly diminishing reliability returns and fast rising design impact (Table 10) is reached after addition of the gyro assembly.

Table 10.   SCS Configuration Tradeoffs

Tradeoff Parameters	SCS Configuration			
	Basic Block II	+GA (standby)	+GA and SDA (AES baseline)	+Gyro and CEA
Mission success reliability (LLE mission)*	0.9733	0.9956	0.9961	0.9974
Δ SCS weight (pounds)	0	21.6	43.1	38.2
Δ SCS volume (cubic inches)	0	580	1603	1292
Δ complexity (total number of interface wires)	0	38	258	111
Contingency fuel (pounds) (for mission success)	< 35	< 35	0	< 35
*SCS mission success reliability goal: 0.9956				

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The addition of the solenoid driver assembly, however, results in more than an increase in reliability. The loss of a jet driver can result in the requirement for additional RCS propellant. However, the potential require- ment for additional RCS contingency propellant, does not appear to justify the addition of the solenoid driver assembly, based on the additional design impact.

All three levels of redundancy appear to meet the reliability goal. These figures may change with changes in predicted failure rates, additions of switching, etc., however, the relationships between the figures and the design impact should remain relatively constant. Therefore, the optimum redundancy choice appears to be the addition of a redundant gyro assembly to the basic Block II SCS.

CREW SAFETY

In addition to mission success requirements, reliability studies were performed to determine the effect of crew safety requirements. The overall AES crew safety goal is 0.999. For the crew safety studies, the combined electronics (GN&C, SCS, and communications) were treated together due to the functional inter-relationship between these systems. The combined electronics crew safety goal is 0.999886.

The most significant aspect of the crew safety studies was to determine the effect of the dormant CMC mode on the communications system. Portions of the communication system are criticality I components (i. e., failures cause loss of personnel) when the dormant clock mode is used. This occurs since portions of the communication system must operate to obtain MSFN navigation data or to start the clock for on-board navigation. Crew safety studies were conducted both with and without continuous CMC standby oper- ation. The results are presented in Table 11.

The communications system baseline is the Block II system. The crew safety studies were based only on the use of the UDL for restarting the clock. The inclusion of the voice link capability will result in a slight improvement and will be evaluated during future studies. Although continuous operation of the CMC standby mode results in improved crew safety, it results in a signifi- cant decrease in the mission success reliability indicated previously. There- fore, it is necessary to consider both mission success and crew safety in determining a system configuration. To achieve both apportionments would require the use of a configuration that involves major redesign and equipment relocation. Further effort is planned to determine modifications to the central timing equipment which could permit completely on-board means for restart- ing the clock and, thus, lessen the requirements that are imposed on the communications subsystem.

Table 11.   Crew Safety for Combined Electronics

Combined Electronics Configuration			Crew Safety
Clock Mode	Communications	G&C	
Operate	Block II	Baseline	0.99983
		Plus redundant PSA (ISS) IMU heaters and clock	0.99995
Dormant	Block II plus redundant criticality I components	Baseline (Configuration A)	0.99970
		Plus redundant PSA (ISS) and IMU heaters	0.99972

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GUIDANCE AND NAVIGATION PERFORMANCE REQUIREMENTS

An analysis was made to determine the performance capabilities of the Block II G&C for AES missions. This analysis was based upon the ground rule that only life extension modifications are acceptable for the G&C flight hardware. Therefore, the purpose of the studies was to determine AES mission functions for which the Block II performance is adequate and to establish the penalties involved where it is not adequate. The penalties expected are additional fuel loads, modifications to the CMC programs, and mission operational restrictions.

Table 2 is a summary of functions requiring further study. Of these, the LPO requirements are expected to be the most demanding because of the significant differences in the trajectory compared with Apollo. Because the LPO return leg represents the greatest differences from Apollo, it was selected for detailed error analysis during the PDP. It is expected that more detailed studies of the other mission differences will be conducted during the FDP.

LPO MISSION RETURN LEG TRAJECTORY

The return trajectory from an LPO mission requires a plane change of approximately 90 degrees because it starts out in the plane of a lunar polar orbit and ends up in a plane near the plane of the moon's orbit. An initial transearth injection (TEI) maneuver of 2190 fps within the plane of the lunar orbit will accomplish this most efficiently. A second maneuver of 3715 fps is made near the sphere of influence of the moon (20,000 nmi from the center of the moon) to accomplish the plane change. Following the plane change maneuver, the remainder of the trajectory (second leg) is similar to the Apollo lunar return. The entry interface is reached 4.1 days after the initial TEI burn at a latitude about 30 degrees south of the equatorial plane and at an altitude of 402,000 feet.

G&C CONFIGURATION

Most of the studies assumed that the worst case G&C configuration was used. This consists of using the onboard GN&C for navigation and the SCS subsystem for control during thrusting. This situation would require two failures, one in the command module and one on the ground (MSFN). The on-board failure would have to occur in the inertial subsystem of the GN&C, since this is the only failure that would necessitate using the SCS for guidance while allowing for on-board navigation. The other failure would have to be in

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MSFN's capability to supply navigation data to the vehicle. Also considered was the configuration for which MSFN provides navigation information and the SCS subsystem is used for control during thrusting. This situation results if the GN&C subsystem fails for any reason (single failure).

LPO MISSION ERROR ANALYSIS

This analysis considered the midcourse corrections required to acheive satisfactory (abort) entry conditions. The return leg of the mission was the only phase covered. The analysis considered all G&C functions performed, beginning with the preparation for TEI and extending through the entry interface.

Because of the large differences between the LPO and Apollo mission trajectories, certain aspects of the real mission criteria are undefined where equivalent situations do not exist between the missions, e.g., the guidance equations that would be used during the portion of the trajectory (first leg) from the TEI to the plane-change burn. A target point for this leg of the trajectory must be defined, and the constraints, which have to be satisfied in arriving at that target, must be specified. Examination of various target constraints has resulted in improvement in correction requirements. However, significant additional improvement should be possible with further optimization of the guidance scheme.

The preliminary results of the study indicated that, without a midcourse correction in the first leg, a delta V requirement of 236 fps (1σ) for the midcourse delta V maneuver during the second TEI burn and a total of 37 fps (1σ) for the second leg must be provided. Because this requirement was considered excessive, it was decided to investigate the effects of including one midcourse correction in the first trajectory leg. Although the inclusion of a fixed time-of-arrival midcourse delta V in the first leg demonstrated an overall savings of midcourse fuel, the requirement was still considered excessive. To further minimize the delta V magnitude, a variable time-of-arrival correction, in which two of the components of target position deviation and one component of velocity were constrained, was used. However, the values of the required delta V closely approximated those of the fixed time-of-arrival philosophy.

The option of correcting two-position components and using the third degree of freedom to minimize the magnitude of the required delta V was included in the variable time-of-arrival program to determine whether the required delta V could be reduced further by removing the constraint that a velocity component be nulled in the midcourse correction. A significant reduction was achieved in the magnitude of the first midcourse delta V. The time selected to apply the midcourse delta V was two hours from the

termination of the injection maneuver. This time was chosen to minimize total midcourse fuel and to allow adequate time for the completion of six navigation sightings.

In previous studies of transearth trajectories with near-equatorial lunar orbit and TEI using the SCS delta V mode, the time of the first midcourse correction has been approximately six to eight hours from injection and the magnitude from 55 to 75 fps, depending on values of parameters used. The midcourse delta V for the present case occurs at two hours, and the magnitude is 82 fps (1σ).

The second maneuver is somewhat longer than the first injection and begins 14. 3 hours after termination of the first burn. The math model used for the first injection cannot be applied to the second burn because of the distance of the vehicle from the moon. The analysis required to evaluate this type of maneuver must use a time history of position, velocity, and acceleration. Since such a time history was not available, it was decided to substitute a powered flight trajectory of similar physical conditions, thrust profile, and total time of burn. The deviations at the initiation of the second TEI burn are used to determine the magnitude of the midcourse delta V that is needed to null the deviations at the entry interface. The pointing accuracy used in calculating this delta V was 1 degree as the correction is part of the injection maneuver. The magnitude of the midcourse delta V calculated for inclusion in the burn was 76. 9 fps (1σ).

The analysis of the second trajectory leg evaluated midcourse delta V's using variable time-of-arrival correction philosophy and calculated the vehicle deviations at the terminal point of the trajectory. Navigation information is based on a total of 50 sightings evenly distributed over the total trajectory time of 83. 78 hours. An optimization procedure was used to determine the star directions providing the greatest reduction in state vector uncertainties along the trajectory.

The analysis of the second trajectory leg was directed toward optimizing midcourse delta V location and minimizing required midcourse fuel using the 20-nmi (3σ) entry corridor targeting constraint. The required delta V for this phase is 12. 6 fps total. For delta V_3 at 8. 26 hours, the magnitude is 7. 4 fps, and the magnitude of a delta V_4 at 55. 8 hours is 5. 2 fps. In calculating the total midcourse delta V for the second leg, the correction during the TEI burn should be included. This correction was applied with the entry corridor as a target condition, as were the two midcourse delta V's given above.

The small values of the delta V's for this leg are due to the application of the midcourse delta V in the injection maneuver. The large delta V in the injection burn compensates for the state vector deviations, and the remaining

corrections are needed to remove the effects of application errors. In this case, the application errors result from hardware inaccuracies and navigation uncertainties.

A summary of the delta V magnitudes and times is given in Table 12 for a corridor constraint of 20 nmi (3σ) using SXT navigation measurements.

Table 12. Summary of Midcourse Delta V's (SXT)

Transearth Trajectory Time (hours)	Number of Navigation Measurements (cumulative)	Delta V Magnitude (fps)
1st TEI 0.0	--	--
Delta V_1 2.0 (1st leg)	6	82.0
Delta V_2 14.28 (2nd TEI)	24	76.9
Delta V_3 22.54 (2nd leg)	29	7.4
Delta V_4 69.9 (2nd leg)	57	5.2
98.06	74	171.5 (1σ)

The previously described study was repeated using MSFN updating (range, range-rate, azimuth, and elevation data) to obtain an estimate of the penalty incurred by using the SXT as the navigation sensor rather than ground-based tracking. Normally, the increased navigation accuracy would reduce the errors in the calculation of the first midcourse delta V. This reduction of errors would not materially affect the magnitude of the first delta V, but its effect would become apparent in the resulting deviations at the target point. Therefore, the midcourse delta V in the second TEI maneuver would be reduced, and the application of this delta V would be more accurate than with SXT navigation, reducing the magnitude of the two midcourse delta V's in the second leg. However, the delta V applied in the first leg did not contain navigation or application errors, and the increased navigation accuracy would not produce the above-mentioned effect. It was expected that the effect of the increased navigation accuracy would be apparent in application of the midcourse corrections in the second leg.

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A comparison of the state vector uncertainties resulting from second TEI burn for the case of MSFN navigation and the uncertainties for the case of SXT navigation shows that any effect of previous navigation is lost in the TEI maneuver. A reduction of approximately three fps is shown in the subsequent analysis of the second leg.

The midcourse delta V magnitudes are as follows: first leg—82 fps, second TEI burn—76.9 fps, and second trajectory leg—9.76 fps. Due to the use of MSFN data, the times of the delta V's in the second leg are altered. The first correction is applied at 15.3 hours and the second at 51 hours. A summary of the delta V magnitudes and times is contained in Table 13 for a corridor constraint of 20 nmi (3σ).

Table 13. Summary of Midcourse Delta V's (MSFN)

Transearth Trajectory Time (hours)	Navigation Update Interval	Delta V Magnitude (fps)
1st TEI 0.0	--	--
Delta V_1 2.0 (1st leg)	Every 1/2 hour	82.0
Delta V_2 14.28 (2nd TEI)	Every 1/2 hour	76.9
Delta V_3 29.58 (2nd leg)	Every 1/2 hour	8.1
Delta V_4 69.3 (2nd leg)	Every 1/2 hour	1.7
98.06		168.7 (1σ)

G&C PERFORMANCE CONCLUSIONS

The results of the analysis of the two-impulse transearth trajectory, using the SCS delta V mode with a 10 arc per second SXT for navigation, show that 171 fps (1σ) for midcourse delta V's is required to satisfy the 20-nmi (3σ) entry corridor constraint. This requirement is a result of executing one midcourse delta V in the first trajectory leg and three midcourse corrections in the second leg. The first midcourse correction in the second leg is included in the plane-change maneuver. All midcourse

corrections are based on the variable time-of-arrival correction policy and they use Apollo subsystem performance values to determine delta V application errors. The results further indicate that 74 SXT navigation measurements are adequate if 24 are contained in the first leg and performed at a uniform rate and 50 are apportioned to the second leg.

The delta V requirement for the SCS delta V mode using MSFN updating is approximately 3 fps (1σ) less than the requirement for the sextant navigation mode. The small savings in required fuel shows that the dominant factor is the execution errors of the delta V maneuvers and that the navigation schedule for optical measurements is adequate.

These requirements are the result of a preliminary tradeoff limited by the study time available. The total delta V requirement established in the analysis does not represent an optimization for the complete trajectory, but is optimized with respect to the separate trajectory legs. Additional study is required in order to determine the interaction of the delta V location and the correction philosophy between the two trajectory legs. Also, the two transearth injection delta V's are impulsive and, as a result, the injection errors calculated can only approximate those with a finite burn.

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CONCLUSIONS

The G&C studies were performed in several steps. The AES transit and housekeeping requirements were translated into G&C operating timelines and reaction control system propellant consumption. The performance and reliability capabilities of the Block II G&C were determined and compared with the AES requirements. As a result, modifications to the Block II were recommended.

The present state of evaluation of the GN&C subsystem suitability for AES requirements provides a configuration concept and some insight into its performance adequacy. The recommended configuration consists of the basic Block II subsystem with the addition of redundancies which will most effectively provide the improved reliability required primarily by the longer AES mission standby operating times. Changes to the Block II configuration consist of the following: (1) CMC clock shutdown when normal operation is not required; (2) addition of a CMC power switch for turning the computer completely off; (3) IMU power switching modification to eliminate suspension power during ISS standby; (4) redundant CDU package; and (5) redundant CDU switching assembly. In addition, it has been shown that a considerable improvement can be obtained by providing redundancy to the heater control circuitry within the IMU, since this circuitry must operate for the full duration of the mission. However, it is not known at this time how much IMU improvement is practical. With no IMU modification, the GN&C reliability for the LPO mission will be 0.983. Depending upon the extent of redundancy provided, IMU improvements could increase the reliability to as much as 0.989. An interim configuration, excluding any IMU improvements, is recommended for future studies. This configuration requires an addition of 0.74 cubic feet and 33 pounds over Block II.

Some portions of AES mission trajectories are more demanding than the Apollo mission on the GN&C performance. All of these mission functions have been identified. The most demanding appear to be the following: (1) plane-change maneuvers for the outbound and return legs of the LPO mission; (2) lunar orbit injection for the LPO mission; (3) orbital navigation for the EPO and LPO missions; and (4) deboost from orbit for EPO and ESO missions. It is not anticipated that any performance improvements will be necessary to the GN&C subsystem, but penalties in the form of additional fuel requirements, computer program changes, and operational restrictions will be involved.

Because the return leg of the LPO mission appears to represent the worst case performance demands on the GN&C, a detailed analysis was performed. The results of this analysis, which assumed the use of backup equipment for both navigation and thrusting, indicate that both additional midcourse correction fuel and substantial CMC program changes from the Apollo lunar mission will be required. The total fuel requirements to compensate for G&C function errors will be 514.5 fps (3σ), as compared with an allocation of 100 fps for the return leg of the Apollo lunar mission. This requirement can be further optimized, but it is apparent that a large fuel penalty will always remain because of the additional implementation errors associated with the two large burns required for transearth injection.

The LLE mission imposes the most stringent reliability requirements on the SCS if this subsystem must provide attitude hold during the lunar orbit astronaut sleep periods. If the requirement for attitude hold during the astronaut sleep period is not imposed on the SCS, the LLE and Apollo LOR mission requirements for the SCS are essentially the same.

The addition of a redundant gyro assembly to the basic Block II SCS is recommended in order to provide the LLE attitude hold capability. The addition of the redundant gyro increases SCS mission success reliability from 0.993 to 0.9956, which can be compared to the desired goal of 0.9956.

If the attitude hold capability is not required of the SCS for the LLE mission, considering the LLE and LPO missions, the following conclusions can be drawn:

1. A basic Block II SCS will provide approximately the same levels of mission success and crew safety probabilities on an LLE mission as are currently predicted for the SCS on an Apollo LOR mission.

2. The basic Block II SCS mission success probability, when used on an LPO mission for transit and housekeeping, exceeds the desired goal of 0.9956.

These conclusions are based on the assumption that the orbital stabilization subsystem will be available to provide attitude hold for housekeeping functions in addition to experiment support. For this case it is recommended that no additional redundancies to the Block II SCS subsystem be provided.

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COMMUNICATION AND DATA SYSTEM

COMMUNICATION AND DATA SYSTEM

The objectives of the study were to determine the minimum modification necessary to the Apollo Block II communications and data (C&D) subsystem to support the AES CSM operational requirements, the definition of any excess capability available for support of experiment activities, and definition of a baseline AES communications and data subsystem for further study.

The study was based upon the Apollo Block II C&D subsystem configuration as of September 1965; this configuration was fully described by the procurement and system specifications existing at that time. Subsequent changes were monitored, but did not affect the course of the study.

The requirements analysis defined certain ground rules which define the operating time and modes of the equipment. These ground rules were established in conjunction with NASA; NASA also postulated certain requirements: (1) CSM/laboratory audio hardline shall be provided, and (2) the laboratory and pallet will provide their own data storage and management systems.

The AES requirements were then compared to the Block II capability. AES C&D modes and equipment utilization profiles for the reference missions were developed. The C&D excess capability was determined. This is the Block II capability which is not required for AES CSM support.

For a more detailed description of the AES communications and data system the reader is referred to NAA/S&ID report SID 65-1521.

AES REQUIREMENTS ANALYSIS

The AES telecommunications requirements are influenced by mission characteristics, orbit trajectory, crew-operation functions, and the ground network configuration, as well as ground support and control requirements, measurement requirements, equipment, and other subsystem design limitations. Four reference missions were considered; their characteristics were determined in the context of transfer of information between spacecraft and MSFN for all phases of the mission.

MISSION GROUND RULES

A minimum functional RF coverage is necessary for the four reference missions. A set of ground rules applicable to the four missions was generated by NAA and NASA to permit subsystem definition:

Tracking

1. Maximum coverage during station contacts between liftoff and insertion into parking orbit

2. Three 5-minute tracks as rapidly as coverage permits to confirm parking orbit

3. One 5-minute track during each succeeding parking orbit

4. During and, for a 5-minute period within 30 minutes, after each thrusting maneuver

5. During each major CSM maneuver (not including routine RCS of orientation maneuvers)

6. Two 5-minute tracks between planetary transfer injection and midcourse correction (approximate even timing assumed)

7. Seven minutes between midcourse correction and correction +15 minutes

8. Five minutes before and five minutes within 60 minutes after planetary approach manuever

9. One 5-minute track for each 8 hours of lunar orbit

10. Three 5-minute tracks as rapidly as coverage permits after initiation of an earth-mission orbit for orbit confirmation

11. One 5-minute track for each 8 hours of earth-mission orbit or position

12. Maximum coverage during station contacts between 5 minutes before CM separation to 400,000 foot reentry

Voice

1. Continuous open receiving channel at CSM to ground network (even when LOS not realized)

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2. Continuous open receiving channel (may be hardline) to experimental module, when inhabited

3. Maximum transmission capability from liftoff through parking orbit

4. Maximum transmission capability from 3 minutes before through 5 minutes after any CSM maneuver requiring astronaut participation or monitoring

5. Maximum transmission capability from 3 minutes before CM separation through crew recovery

6. Minimum of one conversation (average number of transmission to be determined) for each 4 hours of routine mission

7. Contingency transmission - 100 hours

Television

No requirement for television transmission for basic mission support

Up-Data Link

Continuous open channel at CSM (even when LOS not realized)

Telemetry

1. Routine monitoring measurements will be sampled for a period sufficient to insure confidence and transmitted once every two or four hours ±20 minutes (basic period of two hours). Transmission will be in real time (RT) if possible.

2. Timed (trend) measurements will be sampled for a period sufficient to insure confidence and transmitted or recorded every 15, 30, or 60 minutes. Recorded data will be transmitted at least once every four hours.

3. Biomedical measurements for each crew member will be sampled once each four hours ±20 minutes and transmitted in RT if possible.

4. There will be continuous telemetry transmission of selected parameters from liftoff to attainment of the first parking orbit.

5. There will be provision for at least 10 minutes of check-out telemetry for support subsystems between each major mission phase. Transmission will be in RT if possible.

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Recovery Aids

     Continuous operation of recovery beacons required from the time the recovery antennas are deployed to the completion of the recovery operation

APPLICATION OF GROUND RULES

     These ground rules were applied to each of the four reference missions to determine the equipment requirements. A detailed time profile for each function of each piece of equipment was constructed to determine the utilization in terms of on-off cycles, total running time, surplus capability, and lost data. Table 14 summarizes these factors for Reference Mission 1 (low-altitude, high-inclination, 45-day earth orbit).

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Table 14. Computer Profile Mode Analysis Summary

Orbits	Mode I				Mode II			
	PM Count	PM Time	FM Count	FM Time	PM Count	PM Time	FM Count	FM Time
1 - 240	50	250 Min.	48	196 Min.	203	406 Min.	199	788 Min.
241 - 480	60	292.4	60	242	180	360	180	764
481 - 720	48	234.2	48	270	138	276	138	780

Orbits	PM Surplus	FM Surplus	PM Deficiency	FM Lost Data
1 - 240	1888.5	1602.9		49.2
241 - 480	1876.2	1634.7	7.6	53.0
481 - 720	2119.1	1701.9	5.8	132.3
	5883.8 Min.	4939.5 Min.	13.4 Min.	234.5 Min.
	98.1 hrs.	82.3 hrs.		3.9 hrs.

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UTILIZATION OF BLOCK II FOR AES MISSIONS

Apollo C&D capabilities and AES C&D requirements were compared, point by point, to determine the ability of Apollo equipment to satisfy AES functions and determine what modifications were needed. As a by-product, certain excess capabilities were defined.

The AES C&D requirements for other than on-station operations will be similar to Apollo requirements and will be satisfied by Block II equipment assigned those functions. AES on-station operations (after achieving a stable, long-period orbit condition) for Missions 2, 3, and 4 dictate the use of S-band communication links, as the VHF, HF, and C-band equipment are range limited to near earth operations.

Additional considerations of versatility, wideband transmission capability, and MSFN S-band equipment availability indicate S-band communication links should also be selected for the on-station AES communication requirements for Reference Mission 1. This will include the CSM to MSFN functions of voice, tracking, and telemetry, and MSFN to CSM functions of voice and up-data.

The Apollo Block II equipment that would be used to implement the AES on-station C&D requirements of the reference missions are shown in bold outline in Figure 1. This equipment is the following: PCM telemetry unit, data storage equipment, S-band transponder, premodulation processor, S-band power amplifier, S-band omni antenna equipment, up-data link, central timing equipment, and audio center.

SELECTION OF AES COMMUNICATIONS AND DATA MODES

Three principal modes were selected for the AES missions and are listed in Table 15. AES Mode I provides for transmitting from CSM to MSFN the functions of voice, real-time PCM telemetry, PRN tracking and ranging, and PCM stored telemtry. AES Mode I also provides for reception of voice, PRN ranging and tracking carrier, and up-data. AES Mode II is identical with Mode I with the exception of omission of the PRN ranging function. AES Mode III is for reception only and fulfills the requirements that the CSM be capable of receiving voice and up-data at all times.
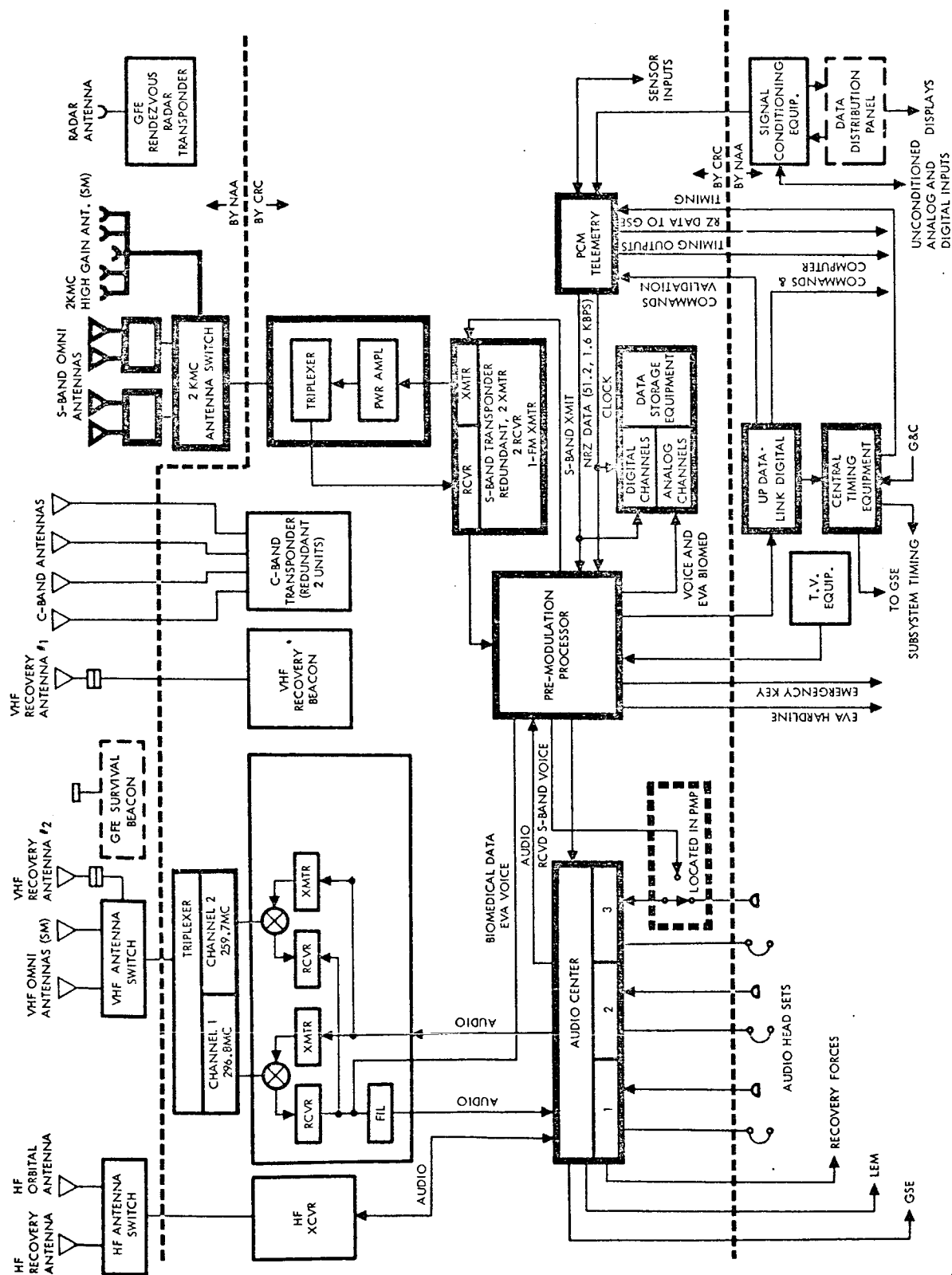
Figure 1. Apollo Block II Communications and Data Functional Block Diagram

Table 15. AES C&D Modes

Mode	Equipment	Information Transfer
I	Transmitters: PM - 2287.5 Real Time Data FM - 2272.5 Stored Data Receiver: 2106.4 Carrier, PRN Voice, Up-Data Link	Real Time Transmission: Duplex Voice, PRN Range/Track, 51.2 kbs Telemetry Playback (1:1) Transmission: stored 51.2 kbs Telemetry Reception: Voice, PRN Range/Track, Up-Data Link
II	Transmitters: PM-2287.5 Real Time Data FM-2272.5 Stored Data Receiver: 2106.4 Carrier, Voice Up-Data Link	Real Time Transmission: Duplex Voice 51.2 kbs Telemetry Playback (1:1) Transmission: Stored 51.2 kbs Telemetry Reception: Voice, up-data link
III	Receiver Equipment Up-Data Link, Voice - 2106.4	Reception: MSFN Voice, Up-Data Link

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UTILIZATION OF BLOCK II C&D SUBSYSTEM FOR AES REFERENCE MISSION 1

A primary difference between the AES and Block II C&D operations is the greater duration of the AES missions (up to 45 days). There are two alternative C&D operation schemes for accomplishment of AES missions: (1) continuous C&D equipment operation throughout the mission as in Block II LOR; or (2) voice reception and UDL on continuously, with the transmitters and associated equipment operating cyclically, for voice, telemetry, and tracking, with on periods only as dictated by specific mission requirements.

Over the 45-day mission, continuous operation increases CSM power consumption by an average of 178 watts, or 14 percent over that required for cyclic operation. In addition, the operating life of the equipment must be extended from 200 hours for the LOR mission to 1080 hours for the AES missions. Cyclic operation will alleviate the power problems, but will introduce the question of the affect upon reliability of on-off switching. On the basis of a preliminary evaluation of these alternatives, the cyclic mode of operation was selected for further study.

Analysis of AES communication mode profiles derived from the communication access tabulation for Reference Mission 1 indicates that the AES communication modes as defined in Table 15 will require on-off cycling and total on-station utilization as shown in Table 16.

Table 16.　Mode Usage-Reference Mission 1
(45 Days, Two Orbits in Polar Earth Orbit)

Mode	Operating Cycle			Total Hours
	Duration (minutes)	Interval (hours)	Number	
I	5	8	158	13.4
II	2 - 5	2	517	38.8
III	Continuous			1080

Results of mode profile derived from communication access tabulation for Reference Mission 1, indicate that these transmission modes, with the continuous voice and UDL receiving capability, satisfy the communication/data requirements of the CSM during Reference Mission 1 on-station phase.

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Figure 2 lists all major functional units of AES C&D equipment involved in on-station operations: it provides a profile, as a function of time in minutes, of the operation of these equipments for Reference Mission 1. The profile diagrammed in Figure 2 uses as abscissa one cycle of equipment operation. This cycle extends over one period of time between initiating of RF transmission in Modes I or II. The solid lines indicate when, during the cycle, each of the equipment units involved must be in its operating mode. The dotted lines pertain to time periods within the cycle when the corresponding units either must be operating or may be turned off depending on certain conditions. The absence of either the solid or dotted line indicates when the corresponding equipment may be switched off. The profile does not take into consideration communications acquisition time or warm-up time requirements or other similar factors.

EQUIPMENT UTILIZATION ANALYSIS FOR REFERENCE MISSIONS 2, 3, AND 4

Reference Mission 1 was chosen as the mission to be studied in detail. However, communications utilization differs between missions. Missions 2, 3, and 4 were analyzed by applying the ground rules previously described. Table 17 is a summary comparison of the equipment utilization by mission.

Equipment Utilization, Reference Mission 2

Reference Mission 2 differs from the high-inclination earth orbital mission in several ways that affect the C&D equipment. If the analysis is constrained to the on-station operation, it is immediately apparent that, since a ground station is continuously available, there is no geometric restriction upon communicating time. Therefore, it is not necessary to record and store housekeeping telemetry, voice, etc., as all these functions can be accomplished in real time.

If the ground rules are applied for this mission, it is seen that transmission of real time telemetry will occur for two minutes every 30 minutes. Hence, the PM transmitter will be turned on 2169 times for two minutes each time, to transmit telemetry during the on-station phase of the mission. Voice transmitting may occur the same number of times as real-time telemetry transmission. Ranging, through the PRN system, requires the use of the S-band transmitter and receiver a total of 135 times for a duration of five minutes each time. Up-voice and up-data link requires that the receiver be on during the whole mission. Table 18 shows the on-station C&D equipment requirements for this mission.

During this mission, the requirement for data storage has been eliminated at the cost of operating the PM transmitter for more cycles and hours.

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MINUTES	0 1 2 3 4 5 6 7 8 9	30η / $30\eta+1$	$30\eta+2$	$30N+3$	-3 -2 -1 0
HIGH-GAIN ANTENNA EQUIPMENT	BACKUP				
OMNI ANTENNA EQUIPMENT	████████████				
TRIPLEXER	████████████				
POWER AMPLIFIER NO. 1	████ ▬ ▬ ▬ ▬				
POWER AMPLIFIER NO. 2	▬ ▬ ▬ ▬ ▬ 2N MINUTES ▬ ▬				
PM TRANSMITTER NO. 1	████ ▬ ▬ ▬ ▬				
PM TRANSMITTER NO. 2	BACKUP				
FM TRANSMITTER	████ ▬ ▬ 2N MINUTES ▬ ▬ ▬				
RECEIVER NO. 1	████████████				
RECEIVER NO. 2	BACKUP				
PMP, TRANSMIT PORTION	████ ▬ ▬ ▬ ▬ ▬ ▬				
PMP, RECEIVE PORTION	████████████				
AUDIO CENTER EQUIPMENT	████████████				
UDL	████████████				
PCM EQUIPMENT	██		2 MINUTES ▬ ▬ N TIMES		
DSE	▬ ▬ ▬ 2.25 N MINUTES ▬ ▬ ▬ ▬		2 MIN N TIMES ▬ ▬ ▬ PLUS 0.25 N MIN		
SCE	████████████				
CTE	████████████				
PRN RANGING CHANNEL NO. 1	APPROX EVERY ████ 4TH CYCLE ONLY				
PRN RANGING CHANNEL NO. 2	BACKUP				
MINUTES	0 1 2 3 4 5 6 7 8 9	30η / $30\eta+1$	$30\eta+2$	$30N+3$	-3 -2 -1 0

KEY

████████ EQUIPMENT IN ITS OPERATING MODE

▬ ▬ ▬ ▬ ▬ EQUIPMENT IN EITHER OPERATING OR NONOPERATING MODE – SEE TEXT

N NUMBER OF RECORDINGS MADE DURING EACH OPERATION CYCLE N = 0,1,2,3,4

η RUNNING PARAMETER SUCH THAT = 1,2 . . . N

Figure 2. Equipment Operation Profile

Table 17. Comparison by Mission of Equipment Cycling and Utilization

Mission Number	1		2		3		4	
Equipment / No. of Cycles and Min.	Number of Cycles	On Hours	Number of Cycles	On Hours	Number of Cycles	On Hours	Number of Cycles	On Hours
USBE PM XMTR and Power Amplifier No. 1	679	32.56	2160	78.75	336	92.4	294	50.4
USBE FM XMTR and Power Amplifier No. 2	673	52.90	-	-	336	28.0	294	9.8
USBE PRN Channel	158	12.93	135	11.25	84	84	42	42
PCM Equipment	2199	73.30	2160	72.0	1344	44.8	672	22.4
DSE	3539	114	-	-	1596	63	1176	22.06
PMP (XMIT Sections)	673	50.66	2160	78.75	504	109.2	378	50.4

Table 18. On-Station C&D Equipment Requirements—Reference Mission 2

AFS Modes	Functions	Number of Times Used	Total Function Time (Hr)	High-Gain Antenna	Omni Antenna	Triplexer	Power Amplifier No. 1	Power Amplifier No. 2	PM Transmitter No. 1	PM Transmitter No. 2	FM Transmitter	Receiver No. 1	Receiver No. 2	PMP Transmit	PMP Receive	Audio Center	UDL	PCM	DSE	SCE	CTE	PRN Ranging	PRN Channel 2
I	Carrier PM	135	11.25	X	*	X	X	*	X	*		X	*										*
I	PRN	135	11.25	X	*	X	X	*	X	*													X	
I	Voice	139	11.25	X	*	X	X	*	X	*				X		X								
I	Real-Time Telemetry	135	4.5	X	*	X	X	*	X	*	•			X				X		X	X		
II	Carrier PM	2025	67.5	X	*	X	X	*	X	*													
II	Voice	2025	67.5	X	*	X	X	*	X	*				X		X								
II	Real-Time Telemetry	2025	67.5	X	*	X	X	*	X	*				X				X		X	X		
III	Voice	Continuous		*	X	X						X	*		X	X							
III	Up-data	(Continuous)		*	X	X						X	*		X		X						

Legend: X = equipment used; * = backup

PM transmitter and power amplifier No. 1: number of ON/OFF cycles—2,160; operating time—78.75 hours

PRN: number of ON/OFF cycles—135; operating time—11.25 hours

PMP: number of ON/OFF cycles—2,160; operating time—78.75 hours

PCM: number of ON/OFF cycles—2,160; operating time—72 hours

Since the FM transmitter and the DSE are not used for CSM housekeeping purposes, they may be deleted from this spacecraft at a weight and power saving or retained for experiment data handling or backup.

Equipment Utilization - Reference Mission 3

Reference Mission 3 involves lunar orbital operations, and thus resembles the Apollo LOR mission except for duration. The lunar orbital period of about two hours limits communication access to MSFN. When these ground rules are applied, it is found that transmission of real time telemetry will occur every two hours for a period of two minutes. Hence, the PM transmitter and PM power amplifier will be turned on and off 336 times for two minutes each time to transmit telemetry during the on-station phase of the mission. The FM transmitter, power amplifier 2, the PMP transmitting portion, and the data storage equipment will be cycled 336 times for six minutes each time in order to transmit the recorded telemetry. Voice transmission may occur coincident with real-time telemetry transmission or 336 times for two minutes each time. The PM receiver and transmitter will be cycled 84 times for a duration of one hour each time for the PRN function. The PCM equipment and the data storage equipment will be cycled 2016 times, two minutes each time, in order to acquire and record telemetry data. The UDL will be on during the whole mission. Table 19 shows the on-station C&D equipment requirements for this mission.

Equipment Utilization - Reference Mission 4

Reference Mission 4 is similar to the Apollo LOR mission except for duration. It is nearly identical to Mission 3 except for the solo-astronaut operation requirement. When the ground rules are applied to Mission 4, it is seen that transmission of real-time telemetry will occur every hour for a period of two minutes, except for every fourth orbit, where it will occur every 30 minutes for a period of two minutes. Hence, the PM transmitter and PM power amplifier will be turned on and off 420 times for two minutes each time to transmit real-time telemetry during the on-station phase of the mission. The FM transmitter, power amplifier 2, the PMP transmit portion, and the data storage equipment will be cycled 294 times for a period of two minutes each time. Voice transmission may occur at the same frequency as real-time telemetry. The PM transmitter and receiver will be cycled 12 times for a period of one hour each time. The PCM and the data storage equipment will be cycled 294 times for a period of two minutes each time. Table 20 shows on-station C&D equipment utilization for this mission.

EQUIPMENT UTILIZATION ANALYSIS

From the computer analysis of MSFN station contact time availability, and the communication profile analysis, a summary tabulation of equipment utilization was prepared (Table 21). This table lists the total number of

Table 19. On-Station C&D Equipment Requirements—Reference Mission 3

AES Modes	Functions	Number of Times Used	Total Function Time (Hr)	High-Gain Antenna	Omni Antenna	Tri-plexer	Power Ampli-fier No. 1	Power Ampli-fier No. 2	PM Trans-mitter No. 1	PM Trans-mitter No. 2	FM Trans-mitter	Receiver No. 1	Receiver No. 2	PMP Trans-mit	PMP Receive	Audio Center	UDL	PCM	DSE	SCE	CTE	PRN Ranging	PRN Channel 2
I	Carrier PM	84	84	X	*	X	X	*	X	*	X												
	Carrier FM	84	2.8	X	*	X	*	X	X	*			*									X	*
	PRN	84	84	X	*	X	X	*	X	*												X	
	Voice	84	84	X	*	X	X	*	X	*				X		X							
	Real-Time Telemetry	252	8.4	X	*	X	X	*	X	*				X				X		X	X		
	Record Telemetry	84	2.8	X	*	X	*	X			X			X				X					
II	Carrier PM	252	8.4	X	*	X	X	*	X	*	X												
	Carrier FM	252	25.2	X	*	X	*	X	X	*												X	
	Voice	252	8.4	X	*	X	X	*	X	*				X		X							
	Real-Time Telemetry	252	8.4	X	*	X	*	*						X				X		X	X		
	Record Telemetry	252	25.2	X	*	X	*	X			X			X				X					
III	Voice	Continuous		X	*	X						X	*		X	X	X							
	Up-data	Continuous		X	*	X						X	*		X		X							
Data Stor-age Opera-tions	Record	840	28.0		*																		
	Rewind	252	3.5																				
	Playback	252	28.0																				
	Rewind	252	3.5																				

See Modes I and II recorded telemetry for equipment used in playback

Legend: X = equipment used; * = backup

PM transmissions and power amplifier No. 1, carrier PM: number of ON/OFF cycles—336; operating time—92.4 hours

FM transmissions and power amplifier No. 2, recorded telemetry: number of ON/OFF cycles—336; operating time—28 hours

PRN: number of ON/OFF cycles—84; operating time—5040 hours

PCM, real-time and record telemetry: number of ON/OFF cycles—252 + 252 + 840 = 1344; operating time—8.4 + 8.4 + 2.8 = 44.8 hours

DSE, record, rewind-playback, rewind: number of ON/OFF cycles—840 + 252 + 252 + 252 = 1590; operating time—28 + 3.5 + 28 + 13.5 = 63 hours

PMP transmit (assume voice and real-time PCM usage occurs with PMP on for recorded PCM transmission): number of ON/OFF cycles—252 + 252 = 504; operating time—84 + 25.2 = 109.2 hours

The following equipment is on continuously: omni or high-gain antenna equipment, triplexer, receiver No. 1, PMP receive, audio center, UDL, CTE, SCE

Table 20. On-Station C&D Equipment Requirements—Reference Mission 4

AES Modes	Functions	Number of Times Used	Total Function Time (Hr)	High-Gain Antenna	Omni Antenna	Tri-plexer	Power Amplifier No. 1	Power Amplifier No. 2	PM Transmitter No. 1	PM Transmitter No. 2	FM Transmitter	Receiver No. 1	Receiver No. 2	PMP Transmit	PMP Receive	Audio Center	UDL	PCM	DSE	SCE	CTE	PRN Ranging	PRN Channel 2
I	Carrier PM	42	42	X	*	X	X	*	X	*													
	Carrier FM	42	84	X	*	X	*	X		*	X												
	PRN	42	42	X	*	X	X	*	X	*		X	*									X	*
	Voice	42	42	X	*	X	X	*	X	*				X		X							
	Real-Time Telemetry	126	41.2	X	*	X	X	*	X	*				X				X		X	X		
	Record Telemetry	42	0.4	X	*	X	*	X		*	X			X				X					
II	Carrier PM	252	8.4		*	X	X	*	X	*													
	Carrier FM	252	8.4	X	*	X	*	X		*	X												
	Voice	252	8.4		*	X	X	*	X	*				X		X							
	Real-Time Telemetry	252	8.4		*	X	X	*	X	*				X				X		X	X		
	Record Telemetry	252	8.4		*	X	*	X		*	X			X				X					
Data Storage Operations	Voice	Continuous		X	*	X					X	*		X	X	X							
	Up-Data	Continuous		X	*	X					X	*		X		X							
	Record	294	9.8																				
	Rewind	294	1.23				See Modes I and II recorded telemetry for equipment used in playback																
	Playback	294	9.8																				
	Rewind	294	1.23																				

Legend: X = equipment used; * = backup

PM transmitter and power amplifier No. 1, carrier PM: number of ON/OFF cycles—294; operating time—50.4 hours

FM transmitter and power amplifier No. 2, recorded telemetry: number of ON/OFF cycles—294; operating time—9.8 hours

PRN: number of ON/OFF cycles—42; operating time—42 hours

PCM real-time and recorded telemetry: number of ON/OFF cycles—126 + 252 + 294 = 672; operating time—9.8 + 8.4 + 4.2 = 22.4 hours

DSE record, rewind-playback, rewind: number of ON/OFF cycles—294 x 4 = 1,176; operating time—9.8 + 1.23 + 9.8 + 1.23 = 22.4 hours

PMP (assume voice and real-time PCM usage occurs with PMP on for recorded PCM transmission): number of ON/OFF cycles—126 + 252 = 378: 4.2 + 8.4 = 50.4 hours

The following equipment is on continuously: omni or high-gain antenna, triplexer, receiver No. 1, PMP receive, audio center, UDL, CTE, SCE

Table 21. On-Station C&D Equipment Requirements—Reference Mission 1

AES Mode	I (158)					II (521)					MSFN to AES III			Data Storage Operation				Summary of Equipment Use	
No. of Transmissions AES to MSFN / Function Transmitted CM to MSFN	Carr. (PM)	Carr (FM)	PRN Voice	Real-Time TLM	Recorded Telemetry	Carr. (PM)	Carr. (FM)	Voice	Real-Time TLM	Recorded Telemetry	Open Recp. Cont.	Voice	Up-Data	Record	Rewind	Playback	Rewind	Total No. ON/OFF Cycles	Total Operation Time (Hours)
Number of Times Used	158	156	158	158	156	521	517	521	521	517	Open	Open	Open	1520	673	673	673		
Total Transmission Time (Hours)	13.46	12.3	12.93	5.27	11.8	19.1	40.6	17.37	17.37	38.86				50.66	6.33	50.66	6.33		
High-Gain Antenna Equipment	*	*	*	*	*	*	*	*	*	*		*	*					*	*
Omni Antenna Equipment	X	X	X	X	X	X	X	X	X	X		X	X					Cont.	1,080
Triplexer	X	X	X	X	X	X	X*	X	X	X		X	X					Cont.	1,080
Power Amplifier No. 1	X	*	X	X	*	X	X*	X	*	*								679	32.56
Power Amplifier No. 2	*	X	*	*	X	*	X	*	X	X								673	52.90
PM Transmitter No. 1	X		X	X		X		X	X									679	32.56
PM Transmitter No. 2	*		*	*		*		*	*									*	*
FM Transmitter		X			X		X			X								673	52.90
Receiver No. 1												X	X					Cont.	1,080
Receiver No. 2												*	*					*	*
PMP Transmit Portion				X	X				X	X		X	X					673	50.66
PMP Receive Portion			X					X				X	X					Cont.	1,080
Audio Center Equipment													X					Cont.	1,080
UDL													X					Cont.	1,080
PCM				X					X					X				2199	73.30
DSE					X					X				X	X		X	3539	114
SCE				X					X					X				Cont.	1,080
CTE				X					X					X				158	12.93
PRN Ranging			X															158	12.93
PRN Ranging Channel No. 2			*															*	*

(See Mode I and II Recorded Telemetry for Equipment Used in Playback)

Legend: * = backup; X = equipment is required for the function listed

PM transmitter power amplifier No. 1, carrier PM: number of ON/OFF cycles—158 + 521 = 679; operating time—13.46 +19.1 = 32.56 hours

FM transmitter power amplifier No. 2, recorded telemetry: number of ON/OFF cycles—156 + 517 = 673; operating time—12.3 +40.6 = 52.90 hours

PRN: number of ON/OFF cycles—158; operating time—12.93 hours

PCM, real-time telemetry and record telemetry: number of ON/OFF cycles—158 + 521 + 1520 = 2199; operating time—5.27 +17.37 + 50.66 = 73.3 hours

DSE, record, rewind-playback, rewind: number of ON/OFF cycles—1520 + 673 + 673 + 673 = 3539; operating time—50.66 + 6.33 + 50.66 + 6.33 = 114 hours

PMP transmit (assume voice and real-time PCM usage occurs with PMP on for recorded PCM transmission): number of ON/OFF cycles—156 + 517 = 673; operating time—11.8 + 38.86 = 50.66 hours

The following equipment is on continuously: omni antenna equipment, triplexer, receiver No. 1, PMP receive, audio center, UDL, SCE, and CTE

transmissions from the CSM to MSFN for each AES communication mode, the C&D functions involved, identification of the individual equipments required to implement the corresponding functions, the number of on-off cycles of each equipment, and the total operating time required per equipment. The impact of implementing these functional and equipment requirements with Block II type equipment is described in the following paragraphs.

PCM Telemetry Equipment

Table 22 shows PCM telemetry requirements based on an analysis of the AES housekeeping measurement requirements.

A comparison of Block II capability with AES telemetry requirements is shown in Table 23. The right-hand column of this table lists the excess or deficient capability of Block II PCM in processing the AES required telemetry. AES will use 29,888 bps of 51,200 bps available leaving an excess of 21,312 bps. On a channel-by-channel comparison Block II will provide an excess capability of two analog channels sampled 200 times per second, 12 analog channels sampled 100 times per second, five channels 50 sps, 66 channels 10 sps, and 24 channels one sps. The excess digital capability is 104 channels (one-bit words) sampled 10 sps.

Table 22.   PCM Telemetry Requirements

Number of Channels	Samples per Second	Bits per Second
ANALOG DATA		
2	200	3200
4	100	3200
10	50	4000
114	10	9120
126	1	1008
DIGITAL OR EVENT DATA		
16	200	3200
144	10	1440
Other Fixed Format		4720
		Total 29,888 Bits per Second

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Table 23.   AES Housekeeping Telemetry Requirements/Capability Analysis

Block II Capability			AES Requirements			Excess or (Deficiency)		
Analog Data			Analog Data			Analog Data		
Chan	S/S	B/Sec.	Chan	S/S	B/Sec.	Chan	S/S	B/Sec.
4	200	6400	2	200	3200	2	200	3200
16	100	12800	4	100	·3200	12	100	9600
15	50	6000	10	50	4000	5	50	2000
180	10	14400	114	10	9120	66	10	5280
150	1	1200	126	1	1008	24	1	192
Digital or Event Data			Digital or Event Data			Digital or Event Data		
Bits	S/S	B/S	Bits	S/S	B/S	Bits	S/S	B/S
16	200	3200	16	200	3200	0	200	–
248	10	2480	144	10	1440	104	10	1040
FIXED DATA		4720	FIXED DATA		4720			
TOTAL = 51,200 Bits/Sec			TOTAL = 29,888 Bits/Sec			EXCESS = 21,312 Bits/Sec		

The in-frame excess of 21,312 bps may be used as acquired and/or transmitted telemetry data interleaved with the 29,888 bps AES housekeeping telemetry. Acquisition and/or transmission of the excess will be coincidental with the acquisition-storage of the housekeeping data. No additional equipment operating time or cycling is involved in utilization of this excess, which requires connection of 0-5 vdc analog signal leads or event signal leads to the PCM telemetry unit commutator.

A summary of AES PCM usage is shown in Table 24. The equipment will be cycled on/off for a total of 2209 times throughout the 45-day mission and will be on for a total time of 76.3 hours. The Block II unit operating life is specified to be 200 hours, continuously operating. The effects of cycling remain to be determined. Block II PCM equipment is suitable on a functional basis for use in the AES C&D subsystem.

Data Storage Equipment

AES C&D ground rules require that the PCM telemetry data be recorded at 30-minute intervals for a period of two minutes each between communication transmission intervals. A Block II type PCM telemetry unit will serve as a source for this data. Recording and playback of the data can be accomplished by using data storage equipment (DSE) similar to that in Apollo Block II.

Table 24.   PCM Telemetry Unit Usage

Usage	Cycles	Hours
Use during data acquisition for storage in DSE	1,520 cycles of 2-minute operations	50.7
Use during real-time data transmission in Modes I and II	679 cycles of 2-minute operations	22.6
Use off-station (other phases)	10 cycles of operation	3.0
Total	2,209	76.3

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Table 21 lists the summary utilization of the DSE for AES Mission 1 on-station operations. Pertinent information is as follows:

	Cycles	Hours
Storage of PCM Data	1520	50.6
Playback of PCM Data	673	50.6
Rewind Cycles	1346	12.6
Total	3539	113.8

A comparison of AES DSE requirements with Block II DSE capabilities indicates that the following features are applicable: The Block II DSE is suitable, on a functional basis, for use in the AES C&D subsystem; however, DSE usage exceeds Block II specified life by a factor of eight, and the effects of cycling (3539 on/off cycles) are not known. Block II excess capacity is record/playback capability for one voice-intercom channel, three scientific analog channels, and one 1.6 K bps PCM channel, usable only in two-minute increments coincident with CSM telemetry recording and playback. Time sharing of DSE telemetry record/playback capability does not appear feasible. The possibility of altering the equipment to playback in the reverse direction should be investigated to reduce the DSE cycling usage.

Audio Equipment

In accordance with the ground rules, the CM is required to have voice reception capability from MSFN continuously available for the full duration of the AES missions. In addition, capability for voice transmission from AES to MSFN must be available a minimum of one conversation for each four hours of routine mission. The CM is also required to have a continuous open duplex voice channel to the experiment module (lab or LEM) when inhabited.

It can be seen from Figure 1 that S-band voice transmission from CSM to MSFN is accomplished by routing signals from the audio center to the PMP (to modulate the 1.25 mc subcarrier), then to the modulator of the S-band PM transmitter. The modulated RF signal is amplified and routed to the antenna equipment for radiation. It can be observed that real time PCM telemetry signals are routed from the PCM unit through the C&D subsystem in a similar manner. Thus, whenever real-time telemetry is being transmitted, all of the C&D equipments required for voice transmission are energized; it is merely necessary to activate the microphone circuits at the audio control panels to obtain voice transmission. For this reason, in the application of AES ground rules for voice transmission, voice-transmission availability was programmed simultaneously with real-time telemetry.

Since the audio center must be energized at all times during the mission (for intercom and voice reception), the transmission of voice from AES to MSFN will impose no additional burden on the C&D subsystem over and above that required for transmission of real-time telemetry.

The Block II audio center is functionally adequate as it is for the AES missions.

Several approaches to the problems associated with experiment module (lab and LEM) voice communications with the CM and the ground were explored. The objective was to provide two-way voice communication between the experiment module and the CM and also between the experiment module and MSFN. All of the solutions studied provide this capability; they differ in the extent to which control autonomy is afforded the experiment-module occupant and, concomitantly, in the extent to which equipment additions/modifications, size-weight-power changes, and reliability compromises are necessary.

The recommended solution does not provide any switch control over the audio equipment from the experiment module. The CM panel switches for a given mode may be preset by the astronaut before leaving the CM for the experiment module; any subsequent changes in these settings must be made from the CM.

The laboratory module is equipped with earphones, a microphone (both are connected in parallel to the existing ones located in the CM), and three receptacles each connected to one CM audio module. The astronaut located in the experiment module may select a CM audio module by inserting his umbilical plug into the appropriate connector. Figure 3 shows a functional block diagram of this solution (the CM audio module is not fully represented).

The voice signal to be transmitted from the experiment module to a ground station is channeled from the microphone to the mike amplifier, to a CM audio module, to the selected transmitter, and to a ground station. Conversely, the signal from the ground station is channeled from the CM receiver to an isolation diode switch circuit, through the earphone amplifier, to the earphone.

Modules 1 and 3 in the Block II CM are equipped with identical components, while Audio Module 2 (navigator module) has an additional automatic switch keyed by a relay. The switch is located in the premodulation processor and is used only with the S-band link. When actuated (Position 2) this switch disconnects the navigator's microphone and connects the one from the experiment module. This position allows the astronaut located in the experiment module to transmit to earth via S-band. When no communication need be relayed between earth and the experiment module, the switch is in Position 1, and the navigator's microphone is in the circuit.

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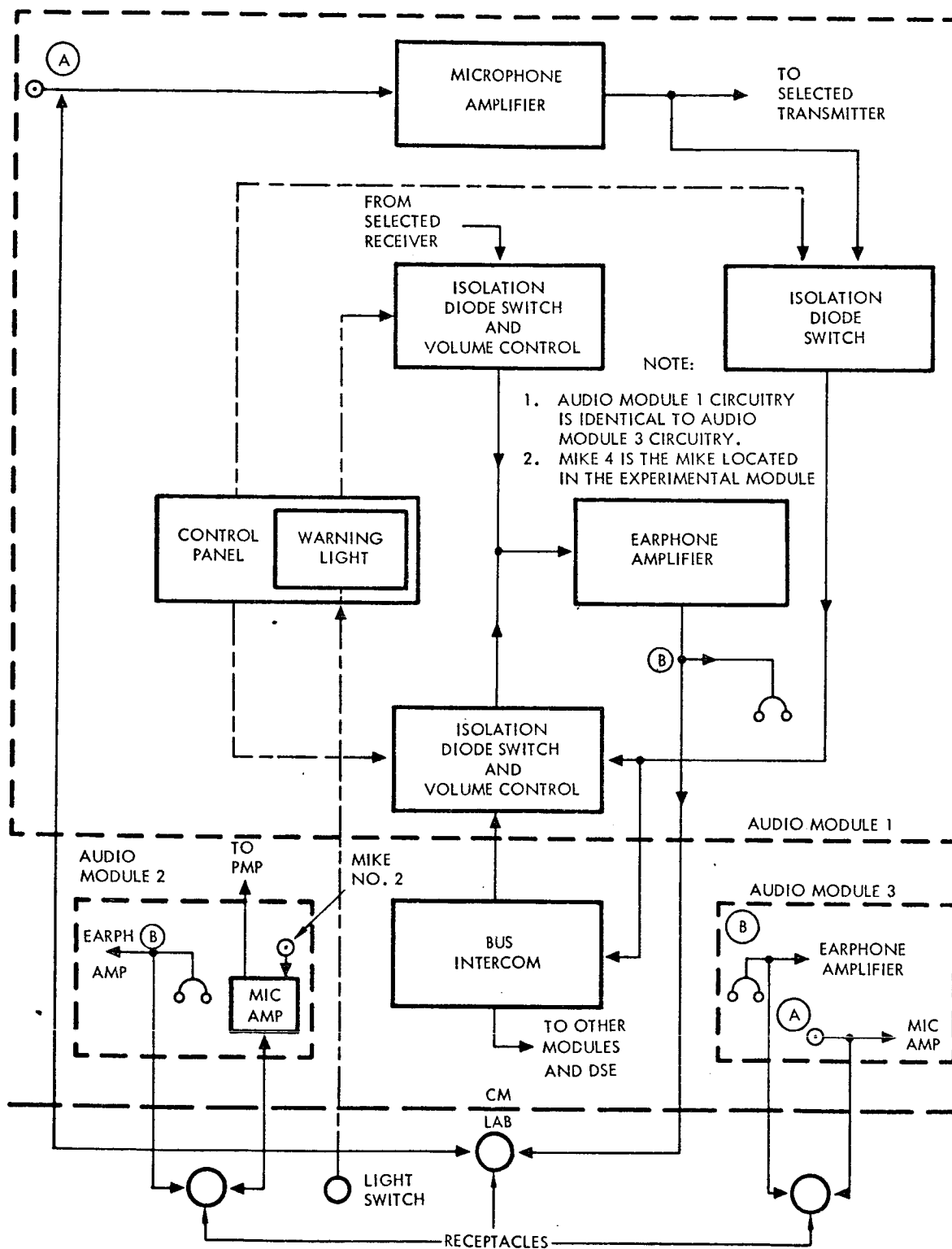
Figure 3. Recommended Experiment Module Audio Link

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The transmitted S-band communications are channeled through the following paths: when the voice initiates in the CM Microphone 2 to microphone amplifier, to switch (Position 1), to unified S-band equipment (USBE); when the voice initiates in the Experiment-Module Microphone 4 to microphone amplifier, to switch (Position 2), to USBE. The transmitted VHF communications are channeled through the following path: Microphone 4 to microphone amplifier to VHF transmitter.

The received S-band communications are handled in the following manner: USBE, to PMP, to switch (Position 2), to Audio Center 2, to selected transmitter (3-way conference mode); or USBE, to PMP, to Audio Center 2, to earphone amplifier, to earphones. HF and VHF-AM receiving communications are handled through normal switching from the control panel in the CM.

Intercommunication between the experiment module and the CM is accomplished by channeling the voice from Microphone 4 to microphone amplifier, to an isolation circuit and diode switch, to intercom bus, to another isolation circuit and diode switch and volume control, to the earphone amplifier, and to the earphones. If the astronaut located in the CM does not turn on his intercom switch, the astronaut located in the experiment module cannot communicate with him. In order to prevent this situation from occurring, a warning light, controlled from the laboratory module, is provided on the main display console.

Up-Data Link

AES C&D ground rules require that a continuous open up-data link (UDL) channel be maintained at the spacecraft, even when line-of-sight communications may not be realized. Hence, the UDL must operate and be receptive to MSFN commands received through the S-band receiver continuously during the AES missions.

As in Block II, the AES digital up-data link is required to receive digital signals from the ground stations, decode and verify them, determine the system which is addressed, and implement commands or route information to the proper destination.

As described in prior sections, Block II UDL provides the capability for addressing eight systems. AES will require five of these addresses; the remaining three: auxiliary decoder word, RTC for external relays, and spare channel, are excess capability.

Real-Time Command Internal Relays

Block II contains 32 double-pole double-throw magnetic latching relays located within the UDL. Immediately after the spacecraft address bits, a

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six-bit octal real-time-command (RTC) word is received. Of 64 possible commands available, 32 command the relays to the "set" position, while the remaining 32 command the relays to "reset."

AES will require 29 of the 32 available relays and 58 of the 64 available commands to implement the control of 14 C&D functions in a total of 43 different modes of operation. The remaining three RTC relays and six octal commands are excess capability.

Guidance Computer

The guidance computer is required to perform similar or the same functions for AES as it is in the Apollo Block II missions. The UDL will be required to recognize and properly route a guidance computer data word-message consisting of a vehicle address word, a system address word, and a guidance computer word. Block II UDL will process such up-data information, and will provide output signals to the AES computer of the proper characteristics such as data rate, pulse width, impedance, etc. Block II UDL will be adequate to implement this requirement.

Central Timing Update

AES will require the capability of up-dating the central timing equipment (CTE) by ground command via the UDL. This function becomes mandatory due to the long mission time involved. Block II UDL provides the capability of up-dating the CTE to a maximum period of 14 days, 23 hours, 59 minutes, and 59 seconds. The AES longer missions will require modification of the equipment to extend the up-data capability to 45 days. Block II UDL will be adequate to implement the timing up-data function with this modification.

UDL Test Message Words

Block II UDL is capable of recognizing and processing two separate consecutive test messages (test messages A and B) for the purpose of bilaterally exercising the UDL logic circuitry to determine its proper operation. This function will be required for AES without change.

Salvo Reset Command

Block II UDL is capable of accepting a salvo command for resetting any one of eight banks of RTC relays, internal or external. AES will require the capability of resetting the internal relays. External relay reset is excess capability.

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Central Timing Equipment

The AES spacecraft requires a continuous source of timing signals for synchronization of various subsystems and functions. The PCM telemetry requires two synchronization signals: a 512,000-pulse-per-second signal and a one-pulse-per-second signal. The PCM telemetry also requires two sets of data inputs, a single-bit signal denoting presence or absence of a CTE synchronizing signal from the CGC, and a 27-bit word which represents accumulated time. The PMP requires a 512-kc signal as a subcarrier frequency for the emergency key mode. The digital event timer requires one 10-pulse-per-second signal. The environmental control system requires a one-pulse-per-ten-minutes timing signal for initiation of water removal from the space suit(s). Table 25 summarizes the AES-required CTE outputs, their destination, and their function. Table 26 describes the Block II destinations and functions of these outputs.

As indicated, four of the CTE outputs are not used by Apollo Block II. These same four outputs are not required by the AES spacecraft. These are three 1-cps outputs and one 512-kcps output, which constitute the CTE excess capability.

There is one discrepancy between the AES requirements and the CTE capability. This is a deficiency in the number of bits output from the time accumulator. There are two bits less than are required; the Block II CTE time accumulator has a 20-day capability, but a 45-day capability is required for AES. There are two solutions to this problem. One is to overlook it and distinguish between two days which are 20 days apart by appropriate means when recording telemetry at the ground station. The other solution is to modify the CTE by adding two one-bit positions to the CTE.

Modifying the CTE will require design effort, incorporation of the modification, and requalification for AES use. Therefore, the first solution is recommended as being the better of the two.

Premodulation Processor

The AES C&D subsystem will require a premodulation processor (PMP) similar to that in Apollo Block II. The Block II PMP is utilized to assist implementation of the following functions: (1) up-link voice detection, (2) up-link digital data detection, (3) relay of up-link S-band voice via VHF-AM transmission (to LEM/EVA), (4) frequency multiplexing for simultaneous transmission of CSM voice and real-time telemetry on the S-band PM transmitter, (5) relay of VHF-AM reception via S-band PM transmission (from LEM/EVA), (6) provision of subcarrier and circuit for emergency key, (7) provision of circuitry for CSM backup voice transmission via baseband on the S-band PM transmitter, (8) auxiliary circuitry for recording CSM-PCM

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Table 25. AES Required CTE Outputs and Destination

Output	Number	Destination & Function
512 kc	2	a. PCM telemetry, synchronization b. premodulation processor, emergency key
6.4 kc	3	3 ac power inverters, synchronization of 400-cycle frequency
10 cps	1	display panel, digital event timer
1 cps	1	PCM telemetry, synchronization
1 pulse/10 min	1	environmental control system, initiate water removal from space suit
Mode monitor	1	PCM telemetry, event measurement
Time Accumulator	27	PCM telemetry, time measurement

Table 26. Block II CTE Outputs and Destination

Output	Number	Destination & Function
512 kc	3	a. PCM telemetry, synchronization b. premodulation processor, emergency key c. spare output
6.4 kc	3	3 ac power inverters, synchronization of 400-cycle frequency
10 cps	1	display panel, digital event timer
1 cps	4	a. PCM telemetry, synchronization b. 3 spare outputs
.1 ppm	1	environmental control system, initiate water removal from space suit
Mode monitor	1	PCM telemetry, event measurement
Time accumulator	25	PCM telemetry, time measurement (digital parallel)

telemetry, CSM intercom voice and voice received from LEM, telemetry received from LEM, and three CSM analog channels of data, (9) frequency multiplexing DSE playback data for transmission via the S-band FM transmitter, and (10) provision for transmission of television or three channels of analog data in real-time via S-band FM transmitter.

AES will require continuous operation of functions 1 and 2. The PMP must be modified to permit the remainder of the PMP equipment to be cycled on/off as required to comply with the communication ground rules. For example, mission number 1 requires 673 on/off cycles with a total utilization time of 3040 minutes to meet the on-station communication requirements. In addition to functions 1 and 2 listed above, AES will also require functions 4, 6, 7, the CSM telemetry portion of 8, and the CSM telemetry portion of 9. Block II PMP excess capabilities are functions 3, 5, all portions of 8 except CSM telemetry, all portions of 9 except CSM telemetry, and 10 in total.

RF Equipment

The RF equipment consists of the unified S-band equipment (USBE) and the S-band power amplifiers. The RF equipment receives processed data from the data equipment and transmits this data through the antenna equipment to the ground stations. The RF equipment also receives up-data, up-voice, carrier, and PRN ranging signal from the ground stations via the antenna equipment. Voice and up-data signals are routed to the data equipment for processing. The carrier is coherently shifted in frequency and retransmitted to the ground for doppler measurement and angle tracking. PRN is reconstructed in the RF equipment and transmitted for ranging.

Unified S-Band Equipment

Block II unified S-band equipment consists of two PM transponders and one FM transmitter with characteristics as follows: The equipment provides communication capability by coherent reception of a 2106.40625 MC carrier phase modulated by a pseudo-random-noise ranging signal, or by a data-modulated 70-KC subcarrier, or by a voice modulated 30-KC subcarrier, or by any and all combinations of these three modulating signals.

The equipment provides communication capability by transmission of a 2287.500-megacycle carrier phase modulated by the received pseudo-random-noise ranging signal, or by a 1024-kilocycle subcarrier, or by a 1250-kilocycle subcarrier or by any and all combinations of these three modulation signals. The equipment provides for simultaneous PM transmission and reception as described above. The equipment is capable of transmission of a 2272.50-megacycle FM carrier. The equipment can provide for FM transmission simultaneous with PM transmission and reception.

In Apollo Block II, provision is made for selecting either of the two PM transmitters for use, but not both simultaneously. Also, there is no provision for operating the receivers without the transponder's transmitter also energized.

AES will require the USBE functional capabilities as described above with the exception of modification necessary to enable separate control of transmission. This modification is required to enable continuous operation of the S-band receive function and on/off cycling of the PM transmitters.

AES Reference Mission 1 will require 1080 hours of continuous PM reception and 679 on/off cycles of PM transmission for a total time period of 32.6 hours. There will be 673 cycles of FM transmission for a total time period of 53.0 hours.

S-Band Power Amplifier Equipment

The S-band power amplifier equipment (S-B PA) interfaces between the communication antenna equipment and the unified S-band equipment. The S-B PA provides amplification for either or both of two S-band transmission channels as required. Two independently controlled amplifiers and associated power supplies are provided, each of which have the capability of amplifying PM or FM signal to a high-power (20 watts) or low-power level (5 watts).

Both channels have filtering to minimize degradation of the C&D subsystem and to avoid propagation of spurious signals. An input from the unified S-band equipment bypasses the normal amplifier to provide direct transmission to the antenna without amplification, when required. In addition, the equipment includes RF transfer switching, control-logic switching, power supplies, and warm-up delay controls.

AES will require the power amplifier capabilities as described. AES mission number 1 will require 679 on/off cycles for power amplifier No. 1 for a total operating time of 32.6 hours, and 673 on/off cycles for power amplifier No. 2 for a total time of 53.0 hours.

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AES C&D SUBSYSTEM EXCESS-CAPABILITY UTILIZATION

The study effort indicated that the Block II equipment can handle all of the AES CSM on-station requirements, with an "excess" available for other purposes. There are three techniques for utilizing the AES C&D subsystem excess capabilities (Figure 4). These techniques are time division, frame division, and frequency division. The time-division technique uses the S-band equipment capabilities during those times when it is not being used for transmission of CSM housekeeping data. The frame-division technique uses the SCE, DSE, and PCM telemetry capabilities not required for CSM housekeeping data. The frequency-division technique uses the unused carriers and subcarriers of the S-band equipment as well as the VHF-AM transceiver concurrent with the transmission of the CSM housekeeping data.

Experimental data may be handled by either the frame-division or the frequency-division technique without any modification except wiring changes. Modifications are necessary for accomplishing the time-division technique.

Utilization of the time-division technique provides experimental data transmission capability equal to the CSM housekeeping data transmission capability. By contrast, both the frame-division and the frequency-division techniques provide limited capability for the handling of experimental data. The frame-division technique is limited by the excess capability contained in the telemetry frame and by the rate at which samples of telemetry data are taken. The frequency-division technique is limited by the capabilities of other data channels which may be operated concurrent with the CSM housekeeping telemetry.

The recommended solution to the problem of handling experimental data is as follows: Use the time-division technique, with an experiment-data handling system time-sharing the communications subsystem with the CSM housekeeping data handling system, for telemetering the experimental data (Figure 5). Use the frame-division technique, which provides for the insertion of data into the CSM housekeeping telemetry frame, for telemetering experiment module housekeeping data. Use the frequency-division technique for experiment data where its special capabilities fit the experiment data requirements.

The provision for continuous operation of the receiving circuits, the UDL, and the CTE, for CSM support permits ground control of experiments whenever the spacecraft is in a receiving position (can be seen by a ground station). Furthermore, a continuous timing signal may be obtained from the
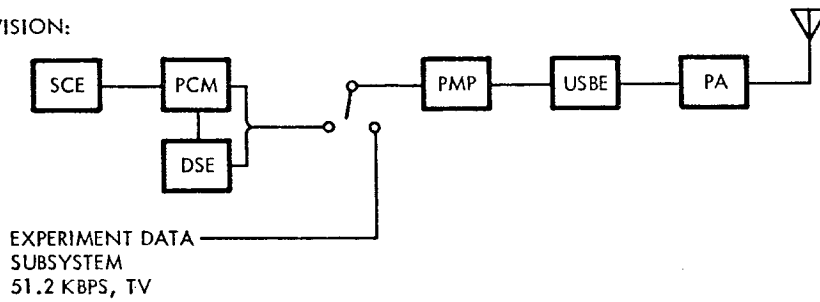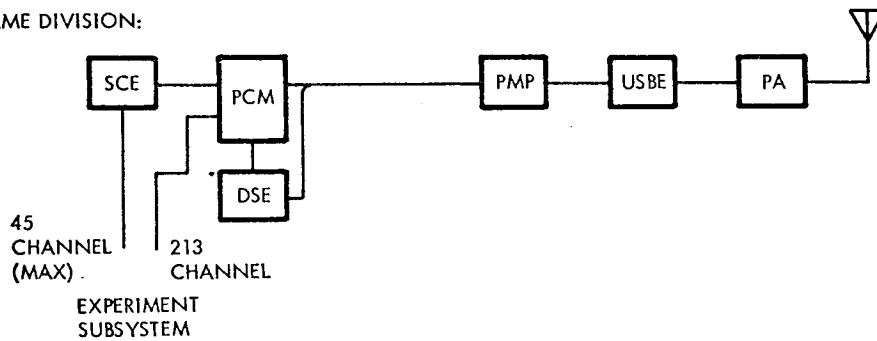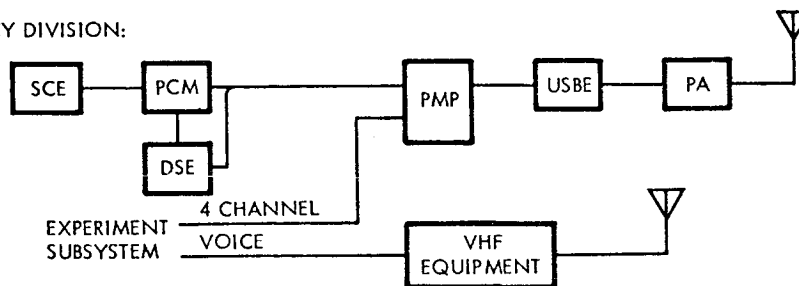
Figure 4.   Excess Utilization Techniques
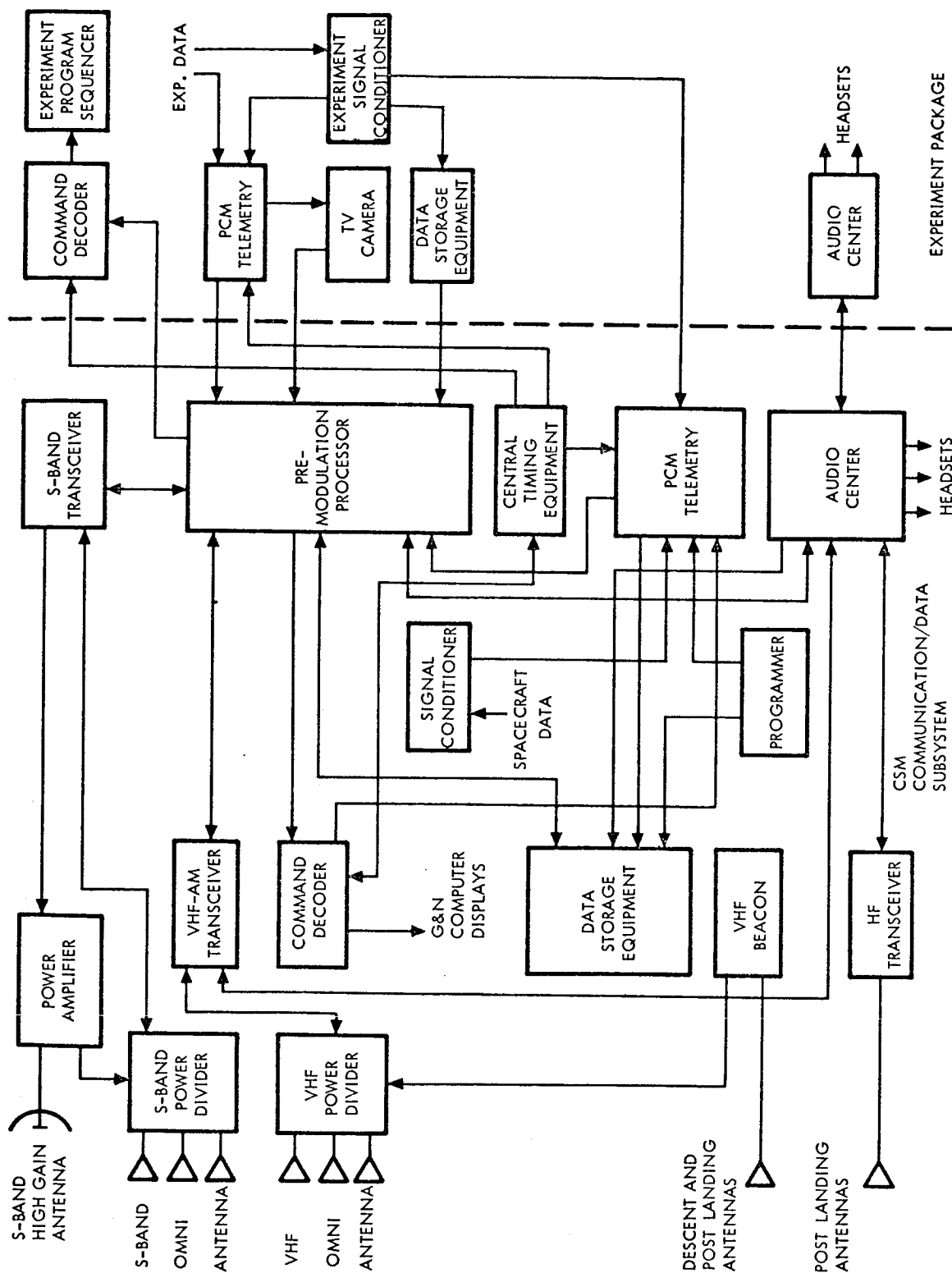
Figure 5. Block Diagram of Time Division Technique

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CTE for synchronizing experiments or for synchronizing the experiment data management subsystem. The provision for cyclic operation of the transmitting portion of the communications subsystem permits, first, the ability to time-share the subsystem, and second, the ability to transmit experiment data only as required by experiment. This ability results in a saving in equipment for the experiment system and a saving in the power charged to experiment use. The experiment system need not supply a communications subsystem, since the CSM subsystem is available for use. The only communications power charged to the experiment system will be that required for operating the transmission portion of the CSM subsystem when it is used for experiment data. Thus, it can be seen that there are significant advantages to the experiment system which may accrue from using the CSM communications subsystem for experiment data transmission and from using the CSM data management subsystem for experiment housekeeping data.

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SUMMARY AND CONCLUSIONS

The detailed C&D functional requirements for the four AES reference missions were analyzed and defined, and the Apollo Block II capabilities were compared with the AES requirements for the purpose of fitting Block II telecommunication equipment capabilities to the AES requirements. Excess capabilities, deficiencies, and required modifications were determined.

To a large extent, Block II equipment will satisfy AES/CSM functional performance requirements. AES C&D requirements do not exceed the Block II functional performance capability although some Block II equipment will be deficient in reliability and operating life due to the extension of the mission time to 45 days (AES maximum). However, the on and off cycling of the C&D equipment as planned for AES has indeterminate effects on reliability. A functional block diagram of the recommended AES C&D subsystem is shown in Figure 1.

The AES C&D requirements for other than on-station operations (voice, tracking, telemetry, and command during the mission phases of ascent, earth parking, preentry, entry, recovery and translunar injection, transearth coast, lunar orbit insertion, lunar orbit, transearth injection, and transearth coast) will be similar to Apollo requirements and will be satisfied functionally by Block II equipment assigned those functions.

Of the C&D subsystem equipment required for the AES-CSM on-station operations, the Block II up-data link equipment, central timing equipment, S-band power amplifier, S-band high-gain antenna, and S-band triplexer require no functional modification for AES utilization. A new programmer is required to control the PCM and DSE equipment during the cyclic recording periods of the PCM telemetry data. The functional differences from Block II and the rationale for the changes are described in the following paragraphs.

The PCM is not functionally different from Block II. In order to perform the cyclic operation schedule for AES, however, it is necessary that the PCM be controllable. The Block II PCM has no on-off control provisions and so circuitry must be added to the AES PCM.

The DSE differs only in its "playback in reverse" capability. The restriction of playback in the forward direction, as in Block II, requires two rewind cycles for each playback cycle. The on-station equipment requirements indicate a requirement for 1346 rewind cycles for Reference Mission 1,

504 for Reference Mission 3, and 588 for Reference Mission 4, if a Block II
DSE is used. With the Block II DSE, data are recorded during one or more
successive recording cycles, the tape is rewound, the recorded data are
reproduced, and the tape is rewound again.

A modification to permit playback in reverse eliminates all of the
rewind cycles. The data may be recorded during one or more successive
recording cycles and may be reproduced while the tape is being rewound.
When all of the data have been reproduced, the tape is again ready for
recording in the forward direction. This kind of operation has the added
advantage that, if any data are lost due to lack of sufficient contact time,
they will be the older data.

The Block II audio center is functionally adequate for AES CSM
operations. The AES requirement that the astronaut in the experimental
module be able to communicate with the CSM and earth by a hardline con-
nection of earphones and microphones to the AES audio equipment was
satisfied. The gain of each of the microphone amplifiers must be increased
to compensate for the signal level decrease due to the parallel connection of
the microphones and the added length of cable. In addition, a warning light
will be added to the CM control panel.

The up-voice and up-data reception functions must be available for use
at any time during the entire AES missions. Therefore, the portions of the
PMP implementing these functions must be continuously activated. The
remainder of the CSM C&D functions are required on an intermittent basis.
The Block II PMP must be modified to provide for continuous operation of
up-voice and up-data demodulators, with separate on/off controls for the
other PMP functions.

The USBE functional requirements are compatible with those of the
Apollo Block II USBE, with the exception that a modification be made to
enable continuous reception and separate control of transmission. This
modification will allow continuous operation of the S-band "receive" function
and on/off cycling of the PM and FM transmitters.

At present within the Block II systems, the S-band omni antenna
switching is accomplished manually as a function of spacecraft attitude.
However, the necessity for numerous switching operations during the 45-day
mission indicates that an automatic switching mode is desirable.

Three additional controls must be added to the AES control panel.
Each of these controls will be a two-position switch. One control will be used
to activate the PM transmitter portion of the selected transponder; one will
be used to activate the transmitting portion of the PMP; and the other will be
used to activate the PCM.

Other controls will be changed in function but will not require additional panel space:

The transponder select switch will be changed to permit independent activation of the transmitter.

The PMP power switch will be changed to permit independent activation of the transmitting portion.

The tape direction switch will be modified to permit playback while the tape is moving in the reverse direction.

The omni antenna select switch will be modified to permit automatic selection of the appropriate omni pair.

A programmer to control the cycling of the PCM and DSE for recording telemetry data is recommended. The astronauts may perform this control operation, but it calls for regular and frequent attention. Furthermore, the lunar landing type missions when a single astronaut is left alone in the space-craft, manual control of this operation would interfere with his sleep period and so an automatic programmer is recommended. The communication functions are not controlled by this programmer, since they do not occur regularly and may be controlled from the ground via the UDL.

The programmer takes the form of a counter to count pulses from the CTE and a set of gates to recognize the count at which telemetry recording should begin and end. The counter will be reset at the end of each recording cycle or whenever telemetry transmission is initiated. Each time a tele-metry transmission occurs, the programmer will begin a sequence of two-minute telemetry recordings separated by 28 minutes of off-time. The length of the sequence is determined by the time between telemetry transmissions.

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INSTRUMENTATION, DISPLAYS, AND CONTROLS SYSTEMS

INSTRUMENTATION, DISPLAYS, AND CONTROLS SYSTEMS

The AES CSM instrumentation subsystem is comprised of the equipment required to monitor the operational CSM housekeeping measurements; that is, those measurements necessary for in-flight management of the spacecraft and preflight check-out. The spacecraft instrumentation subsystem consists of (1) various types of sensors for converting physical and electrical phenomena into electrical signals; (2) signal conditioners for conditioning the electrical signals to proper values for input to the various data utilization subsystems (that is, communication, displays and controls, and ground support equipment); and (3) the system that distributes signals to the various data utilization subsystems.

The AES CSM display and control subsystem includes the main display console and the auxiliary panels used to provide information and control accessibility to the crew for monitoring and management of the spacecraft housekeeping operations. The subsystem consists of meters, display/readout devices, and various switches and control associated with other spacecraft subsystems; accessory signal processing and control logic circuitry; and the spacecraft cabling that connects these components, as well as the interface between the crew and the spacecraft systems.

Instrumentation, displays, and controls are directly dependent upon other spacecraft system requirements and characteristics. Consequently, the detail of the measurements must be considered only as representative for each subsystem since detailed definition of these other subsystems will occur in future studies. More detailed analyses and the AES Instrumentation Measurement and Equipment list and the AES Master Displays and Controls list are contained in SID 65-1522.

INSTRUMENTATION SUBSYSTEM

The AES spacecraft instrumentation subsystem is comprised of the equipment required to monitor the CSM performance, including those measurements necessary for in-flight management of the spacecraft systems and for preflight check-out. The spacecraft instrumentation system consists of (1) various types of sensors for converting physical and electrical phenomena into electrical signals; (2) signal conditioners for conditioning the electrical signals to proper values for input to the various data utilization

subsystems (that is, communications, displays and controls, and ground
support equipment); and (3) the J-box which distributes signals to the various
data utilization subsystems.

The AES instrumentation subsystem requirements are based on those
applied to the Apollo Block II (Spacecraft 101) instrumentation subsystem.
The only deviations from Block II are those that result from spacecraft sub-
system changes attendant with the extended mission duration and variations
in operational procedures.

AES INSTRUMENTATION REQUIREMENTS

Instrumentation operational life (transducer reliability) should be vali-
dated for the longer duration of the AES missions. The safety of the crew
and the management of spacecraft supplies are critical, and the effect of
incorrect measurement data can be disastrous. The lifetime requirement
for transducers will be incorporated in the component procurement specifica-
tion as follows: all instrumentation equipment should be capable of continuous
operation for at least 1200 hours without any deterioration in performance
and 2600 hours total intermittent operation. The 1200-hour operation capa-
bility is based on the AES 45-day mission requirement plus pad operation,
and the 2600-hour intermittent operation includes the time established by
Apollo Block II for calibration, system check-out, standby, countdown, and
mission requirements (1200 hours for mission, 1400 hours for check-out).

Operational procedures (changes in subsystem performance or equip-
ment) dictate different measurements, and changes made in the instrumentation
subsystem are directly dependent upon the other spacecraft systems. Every
subsystem designer and the operations planners were contacted to determine
the measurement requirements, using the Block II measurement list as a
baseline. The information was used to prepare a preliminary AES measure-
ment list. Table 27 contains a summary of the types and number of flight
operational measurements. Table 28 is a measurement destination summary
for the operational measurement list.

AES INSTRUMENTATION SUBSYSTEM DESCRIPTION

The AES instrumentation system block diagram is shown in Figure 6.
A total of 795 measurements is required for housekeeping in support of the
AES missions. The numbers shown in parentheses indicate previous Block II
instrumentation requirements.

Excess capability in the instrumentation subsystem is defined as access
to the measurement system through the instrumentation J-box. The instru-
mentation J-box will be rewired to meet the AES requirements, and access
to the signal conditioning excess capabilities and inputs for the telemetry

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Sub
Structures
Electrical Power
Master Events Sec
Earth Landing Sec
Environmental Cont
Guidance and Navi
Stabilization and
Life Systems
Flight Technolog
Service Propulsion
Reaction Control
Crew Safety
Communications and
TOTAL

System	Phase	Current	Power	Freq.	Pos.	Bio-med	Rad.	
		13		2				
...nce Controller								
...ence Controller								
...trol								
...ation	3							
...ontrol					7			
						3		
...stem					8			
								22
					1			
...entation			2	1	2		8	
	3	13	2	3	18	3	8	72

Table 27.   Operational Flight Measurements Summary

Press	Quan	Rate	Temp	Volt	Time	Event	pH-Acid.	Accel.	Total
			10						10
27	6	8	12	21		26	1		116
				5		8			13
1									1
11	4	1	6			2			24
			1	22		18			44
		4		3		35		1	50
						3			6
					6	1			7
6	7		2			10			33
22	4		6			26			58
5			1			27			34
			1	16		8			38
72	21	13	39	67	6	164	1	1	434

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Table 28.　Measurement Destination Summary

System	Number of Functions					
	PCM*	PCME*	PCMD*	DISP*	GSE ACE*	System Total**
Structures	10	0	0	0	0	10
Electrical Power	67	5	0	111	49	147
Master Event Sequence Controller	2	11	0	0	89	102
Earth Landing Sequence Controller	0	0	0	1	32	33
Environmental Control	15	0	0	22	0	25
Guidance and Navigation	25	4	1	15	78	118
Stabilization and Control	8	28	0	22	38	88
Life Support Systems	3	3	0	0	0	12
Flight Technology	0	0	0	7	0	7
Service Propulsion	15	0	0	25	58	71
Reaction Control	26	0	4	58	53	94
Crew Safety	2	6	0	30	0	33
Communication and Instrumentation	26	3	3	10	30	55
Total	199	60	8	301	427	795

*PCM:　Operational telemetry analog
　PCME:　Operational telemetry event
　PCMD:　Operational telemetry digital
　DISP:　Displays and control

**Totals shown represent the total subsystem measurements made and do not represent a horizontal total as the same measurement may be telemetered, displayed, and/or presented to GSE.

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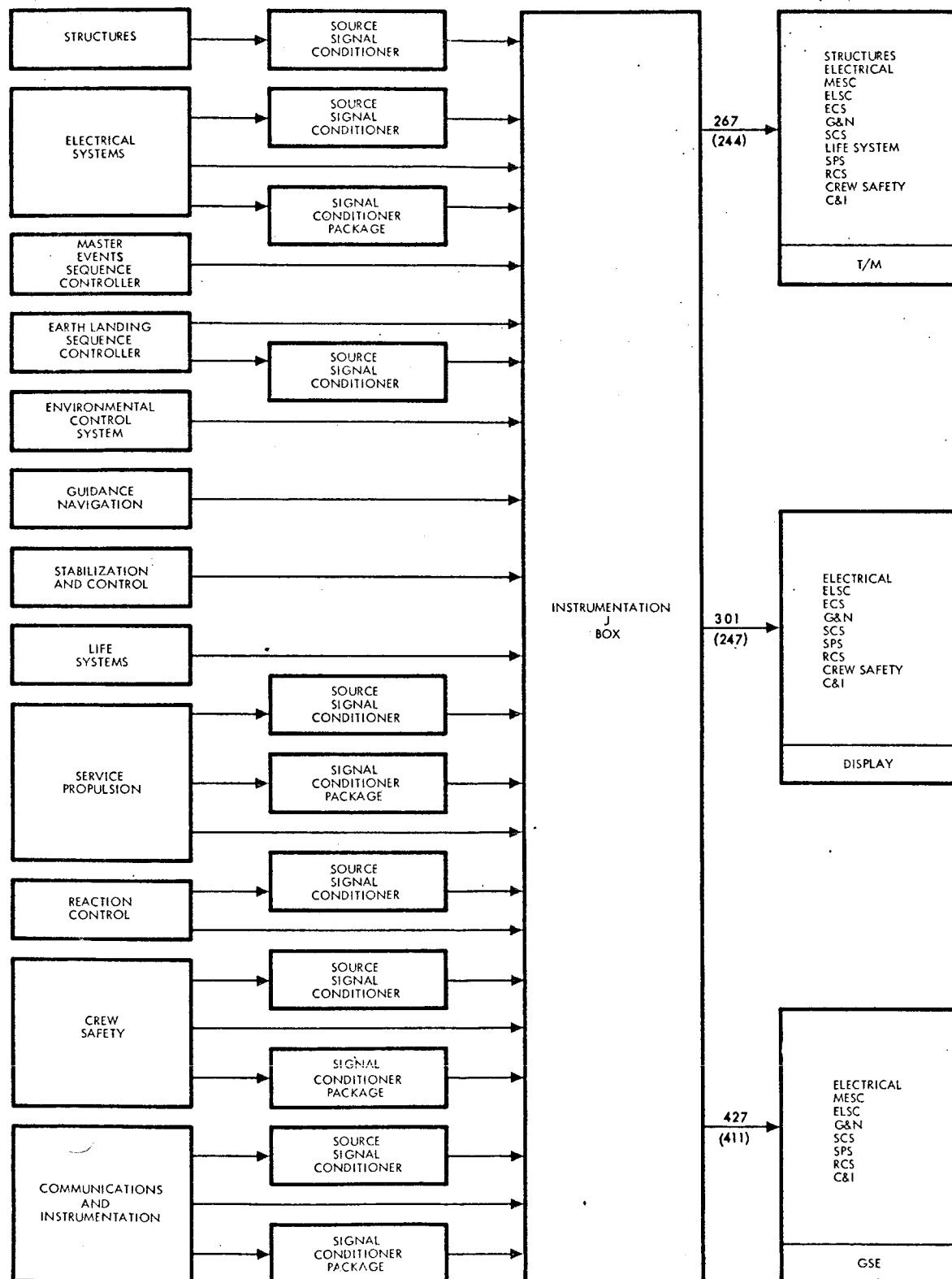
Figure 6. Measurement System Block Diagram

excess channels could be included in the rework. The J-box is located in the lower equipment bay and is best described as a plug-in box with 22 100-pin connectors. Cross-wiring in the J-box base provides the termination and the routing of instrumentation measurements to the proper destination. The J-box connectors may be used for programming experimental measurements into the excess capabilities of the signal conditioners and C&D subsystem. Excess capability of the signal conditioning equipment exists in terms of 10 unused sockets for a variety of plug-in conditioning modules and other modules that are not completely utilized.

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DISPLAYS AND CONTROLS SUBSYSTEM

The functions provided by the displays and controls (D&C) equipment should provide the flight crew with sufficient depth of information and command access to the spacecraft systems to enable them to accomplish the following certain decisions or actions: (1) manual CSM system operation as required under normal mission conditions and contingency operations; (2) safe shutdown of the CSM equipment; (3) monitoring of CSM system equipment as required for normal mission or contingency operations; (4) recognition of malfunctions or incipient hazards to the crew, the vehicle, or the mission in operating the CSM systems, and effecting adjustment or selection of alternate system elements or changes if normal system operation cannot be restored by any of the above actions; and (5) monitoring of CSM system condition and CSM propellant reserves and energy sources as necessary for normal or contingency operations.

Studies conducted to determine operating conditions, interface requirements, and design criteria of the displays and controls subsystems revealed that the majority of the components used in the displays and controls subsystem for the Apollo Block II CSM would also satisfy the AES requirements. A possible exception exists in the area of life expectancy. The D&C subsystem changes that have been requested result from changes occurring within other spacecraft systems.

AES DISPLAY/CONTROL REQUIREMENTS

The design criteria for the AES displays and controls subsystems were the same as those established for the Apollo Block II program (SID 64-1345). The AES extended mission requirements are not expected to compromise these criteria.

The location and arrangement of the display and control panels are identical to Block II. Components have been rearranged on the panels to accept the increased requirements of the subsystems. These changes have been principally in the areas of system management. Reaction and control and environmental control subsystem modifications require additional monitoring and control functions. The addition of the fourth fuel cell and in-space startup capability and their monitoring and control requirements will affect the panel layout.

The increased monitoring and control requirements of the AES subsystems have increased the caution and warning display requirements beyond

the Block II capability. Revisions to the caution/warning system will accommodate approximately an additional 30 comparators and 7 lamp drivers in support of the expanded AES subsystems. The annunciator display will be modified to include the additional warning elements. Final details and modification requirements will be deferred until the Final Definition Phase.

A summary by subsystem of the display and control requirements for AES is presented in Table 29. The numbers shown in parentheses refer to corresponding Block II requirements.

Table 29. Display and Control Signals

Subsystem or Function	Displays	Caution and Warning	Controls
EPS	112 (90)	53 (44)	60 (52)
ELS	1 (1)	—	20 (20)
ECS	25 (22)	11 (8)	21 (20)
SCS	25 (15)	10 (2)	40 (40)
SPS	20 (22)	9 (8)	16 (16)
C&D	11 (9)	1 (1)	75 (75)
Flight Technology	7 (6)	—	6 (6)
RCS	58 (43)	24 (16)	32 (20)
Crew Safety	33 (33)	1 (1)	19 (19)
G&N	4* (4*)	3 (3)	7 (7)
MESC	2 (2)	—	8 (8)
Lighting	0 (0)		17 (17)
Docking	0 (0)	1 (0)	6 (6)
Experiments	0 (0)	1 (0)	0 (0)
Total	298 (247)	114 (83)	327 (306)

Notes: Values outside parentheses are for AES; parenthetical values are for Apollo Block II

*1 DSKY, 3 controls

AES DISPLAY/CONTROL SUBSYSTEM DESCRIPTION

The AES displays and controls subsystem provides the spacecraft flight crew with the information and access necessary to accomplish the various mission objectives. A preliminary definition of the displays and controls subsystem was made in order to determine the effects that the AES mission would have on the Apollo Block II vehicle. As a result of the study effort, a detailed Master Displays and Controls List with a Master Layout Drawing was developed.

The location of the display and control panels is the same as that selected for Block II. The main display console (MDC) is located above the crew couches. Secondary panels, the right-hand and left-hand side display consoles (referred to collectively as the SDC and individually as the RSDC and LSDC), are located adjacent to the right- and left-hand edges of the MDC and on the arm rests of the left-hand and right-hand crew couches. Other locations for D&C equipment are in the left-hand forward equipment bay, right-hand forward equipment bay, and in the navigation station at the lower equipment bay.

In both areas, the display and control functions have remained the same, but the number and/or type of components has increased. The changes to the RCS D&C group have increased the number of switches required to functionally operate the expanded AES RCS. Similarly, the ECS group has been expanded to provide display and control of cryogenic nitrogen storage used for the two-gas cabin atmosphere. To retain the general subsystem grouping stipulated for Apollo Block II without increasing the panel area requirements, it was necessary to postulate a new meter unit. This meter unit will be an extension of the present meter to include a four-movement or quad package. Preliminary investigation has indicated that the package will occupy roughly three-fourths of the panel surface area required for two dual-meter packages. Full definition of the meter specifications will be developed during the Final Definition Phase of the AES study.

In the panel area classified "other," only one subsystem has developed specific design changes affecting the displays and controls subsystem. The electrical power subsystem (EPS) has been revised to include a fourth fuel cell and provide the capability of in-flight startup of fuel cells. In-flight startup details will not be firm until the Final Definition Phase and, therefore, are not included in either the Master D&C List or the preliminary panel layout. The addition of the fourth fuel cell increases the number of displays and controls needed to operate the fuel cells but does not change the approach or function. The above subsystem requirement changes have required relocating components from the main display panel to the auxiliary panels. Additional changes on the auxiliary panel were made by adding controls and circuit breakers in support of the changes required by the subsystems.

No excess capabilities exist on the proposed AES display and control panels. Additional space for D&C could be made available, perhaps in the lower equipment bay. Two areas that would merit further investigation are plug-in panels for experiments and an integrated display and control approach.

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ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEMS

ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEMS

The environmental control system (ECS) and life support system (LSS) equipment comprise those vehicle subsystems that provide a livable environment for the crew, maintaining atmosphere composition and purity, temperature, ventilation, spacecraft and equipment temperatures, and providing sustenance and personal care.

The approach taken for the technical analyses was to evaluate the Block II ECS and LSS for applicability to the AES missions, and then develop required modifications where the former systems were inadequate. Trade-off studies were performed, and a final recommended AES system was developed.

In general, the evaluation and studies show that the basic Block II ECS and LSS concept is adequate for AES, with respect to the methods of control. The AES extended mission duration has affected certain subsystems, such as CO_2 removal, so that a regenerable method has been selected rather than the expendable one now in use.

The ECS study was divided into various phases. First, AES-ECS requirements and constraints were established and the capability of the Block II ECS determined. Next, each major portion of the ECS was considered, its adequacy for AES determined, and trade-offs performed to establish the modifications required to meet AES requirements. The various portions of ECS treated were: (1) atmosphere supply and control, (2) CO_2 control, (3) water separation, (4) suit circuit, (5) atmosphere interchange, (6) trace contaminant control, (7) cold plates, (8) radiators, (9) water management, and (10) cabin temperature control.

The life support provisions for Apollo Block II were evaluated, and a comparison was made with the AES requirements. In general, the increased AES mission duration affects only the quantities of expendables such as food, cleansing pads, etc., without requiring changes in the equipment used to store and handle them. Improved methods for body washing were investigated.

A more detailed description of the ECS and LSS studies is presented in SID 65-1523, the environmental control and life support system document of this report.

ECS REQUIREMENTS

Basic ECS requirements are the same as Block II and include provisions for a conditioned atmosphere within the CM for the crew and thermal control of certain equipment consistent with mission requirements. Water management, provisions for charging the portable life support system (PLSS), and postlanding ventilation are included in the ECS functions. Four basic reference missions were used as a study base. The Block II ECS was evaluated, and modified as required, to accomplish all of the four reference missions.

Significant mission parameters different from Block II are:

1. All AES missions have a longer duration (20 to 45 days), compared to the Block II 14-day mission. The longer duration affects expendables, contaminant removal systems, and reliability.

2. Some reference missions present the spacecraft with a different space environment from Block II. This impacts the cabin temperature control and the radiator systems. One reference mission (LEM taxi), is similar to Block II except for the duration.

3. Electronic equipment operating timelines are different from Block II. Inasmuch as the electronic cooling (cold plate system) is a part of the ECS, as well as its source of heat, the electrical loads resulting from the AES timeline analysis have an affect on the entire ECS.

Additional constraints and/or ground rules were used in the evaluation as follows:

1. Requirement for study of a two-gas CM atmosphere consisting of 70 percent oxygen and 30 percent diluent, with oxygen partial pressure controlled to 3.5 psia. This compares to the Block II 5.0 psi pure oxygen atmosphere.

2. A trade-off study was required between various CO_2 removal concepts, based on the molecular sieve, and the existing Block II lithium hydroxide system.

3. A total of three crewmen variously located between the CM and LEM were considered for thermal analysis purposes.

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4.   The LEM was considered thermally independent from the CSM. Atmosphere gas mixture circulation between CM and LEM was evaluated. Leakage and repressurization requirements were based on requirements of both the CSM and LEM.

5.   No water transfer to the LEM was considered in the analyses.

6.   Only one ECS configuration was considered to accommodate all four design reference missions. Conditions affecting system design and/or performance were used from the mission providing the most severe requirements.

7.   No thermal provisions for experiments were included.

8.   Abort time used for thermal and water management studies was 108 hours to earth return from lunar orbit. Abort time from earth orbit was 6 hours maximum.

The thermal environment of the spacecraft influences the ECS in two areas: (1) heat lost or gained through the CM structure which affects the overall heat balance as well as cabin temperature, and (2) the combination of sun, space and earth/moon that the radiator "sees." This thermal environment was defined for various representative phases of all AES missions for evaluation of the ECS heat balance, cabin temperature effects, and performance of the radiator.

Increased mission duration also has two basic effects on the ECS; reliability and contaminant control. Reliability of the ECS is discussed in report SID 65-1535, and was not a major consideration in the ECS studies. The longer AES missions required a trade-off study regarding CO_2 removal equipment between use of nonregenerable systems as used in Block II, and regenerable types which had previously been narrowed down to the molecular sieve concept.

All ECS subsystems were individually evaluated for performance considering the AES requirements. These subsystems are comprised of (1) cabin atmosphere control, (2) suit circuit and molecular sieve integration, (3) water management, (4) cold plates and network, and (5) radiator system.

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ATMOSPHERE SUPPLY AND CONTROL

One of the requirements of the study was to investigate the use of a two-gas atmosphere as compared to the Block II pure oxygen atmosphere. The use of nitrogen gas as the diluent was considered, although at a later point in time, the use of helium was suggested. A trade-off study of the one- and two-gas systems, based on factors of physiology, weight, reliability, and system integration was performed.

TRADE-OFF STUDY

The data on physiological effects of prolonged confinement in atmospheres of pure oxygen, and mixtures of oxygen and a diluent gas, at reduced total pressures, are neither plentiful nor conclusive. Several tests have been carried out using such atmospheres for periods up to 30 days with no real adverse effects, nor definite recommendations as to the superiority of one over the others. An objection to the pure oxygen atmosphere at 5 psia is that the oxygen pressure exceeds the sea-level normal of 3.1 psia, and can cause pulmonary and aural atelectasis. These and other symptoms have been observed during tests, but in some cases disappeared with acclimitization. A two-gas atmosphere would eliminate these problems.

The primary problem with the two-gas atmosphere is associated with the "bends" or formation of nitrogen bubbles in the blood; this might occur when changing from shirtsleeve operation at 5 psia to emergency suit operation at a pure oxygen pressure of 3.5 psia. However, acclimitization studies have shown that the most severe effects of denitrogenation can be eliminated, that the change in barometric pressure from 5 to 3.5 psia is tolerable, and that the crew should not experience decompression sickness.

The other advantages of using a diluent are that the fire hazard associated with pure oxygen is reduced, and, with helium, the increased heat capacity provides higher capacity for convective heat transfer. The two-gas system requires additional components listed below:

Item	Weight (Pounds)
O_2 partial pressure	1.0
O_2 partial pressure controller	1.0
O_2 flow control valves	0.4
High pressure check valves	0.2
Flow limiter	0.1
Shutoff valve	0.1
N_2 pressure regulator	1.0
	3.8

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The trade-off study originally considered the two-gas system to be at 7 psia, at which pressure the leak rate is higher, requiring an increase in cryogenic stores and tank weight. However, the two-gas system at 5 psia has the same leak rate as pure oxygen at 5 psia, so there is no leakage weight penalty for the two-gas system over the one-gas system. The additional components required may be considered negligible in weight.

The addition of the diluent gas requires tankage and attendant plumbing and the components listed above, as shown in the recommended ECS schematic, Figure 20. The system is easily arranged to afford operation in the one-gas or two-gas mode. Because of the lack of clear-cut data on physiological factors, and that other trade-off factors do not show marked advantages for one or the other, it was decided that for AES, the atmosphere supply system should be such that both a one- and two-gas system can be used.

The atmosphere supply and control system for AES is an automatic system with provisions for normal selection of either a 5 psia pure oxygen or a 5 psia oxygen-diluent atmosphere. The system automatically provides a 3.5 psia pure oxygen suit loop atmosphere during planned or emergency cabin depressurization.

SYSTEM CHARACTERISTICS

The atmosphere supply and control system for the AES/ECS is shown in Figure 7. For operation at 5 psia with pure oxygen, the diluent supply is shut off by closing the diluent shutoff valve. The oxygen supply enters the CM when the oxygen shutoff valve is opened. The cryogenic oxygen supply pressure is reduced to 100 psia by a pressure reducer. Oxygen then flows to the cabin pressure regulator which controls the total cabin pressure to 5 psia pure oxygen. The oxygen partial pressure control system can be deactivated for this mode of operation by a switch in the partial pressure control shown in Figure 8; this is not necessary, however, since the partial pressure of oxygen is above the 3.5 psia set point of the partial pressure sensor. Thus, the sensor, control, and valve will be in a standby mode. The emergency inflow control valve supplies oxygen to the cabin to maintain the pressure above 3.5 psia in the event of a meteoroid penetration of the cabin wall. When the cabin pressure falls below 3.5 psia, the cabin pressure regulator and cabin emergency inflow control valve shut off automatically, and the suit demand regulator maintains the suit circuit pressure at 3.5 psia.

When a two-gas atmosphere is desired, the cabin regulator of the oxygen system is deactivated by the closing of the oxygen shutoff valve upstream of the regulator and the opening of the corresponding shutoff valve on the diluent supply. The oxygen partial pressure system is activated through the oxygen partial pressure control. The oxygen partial pressure
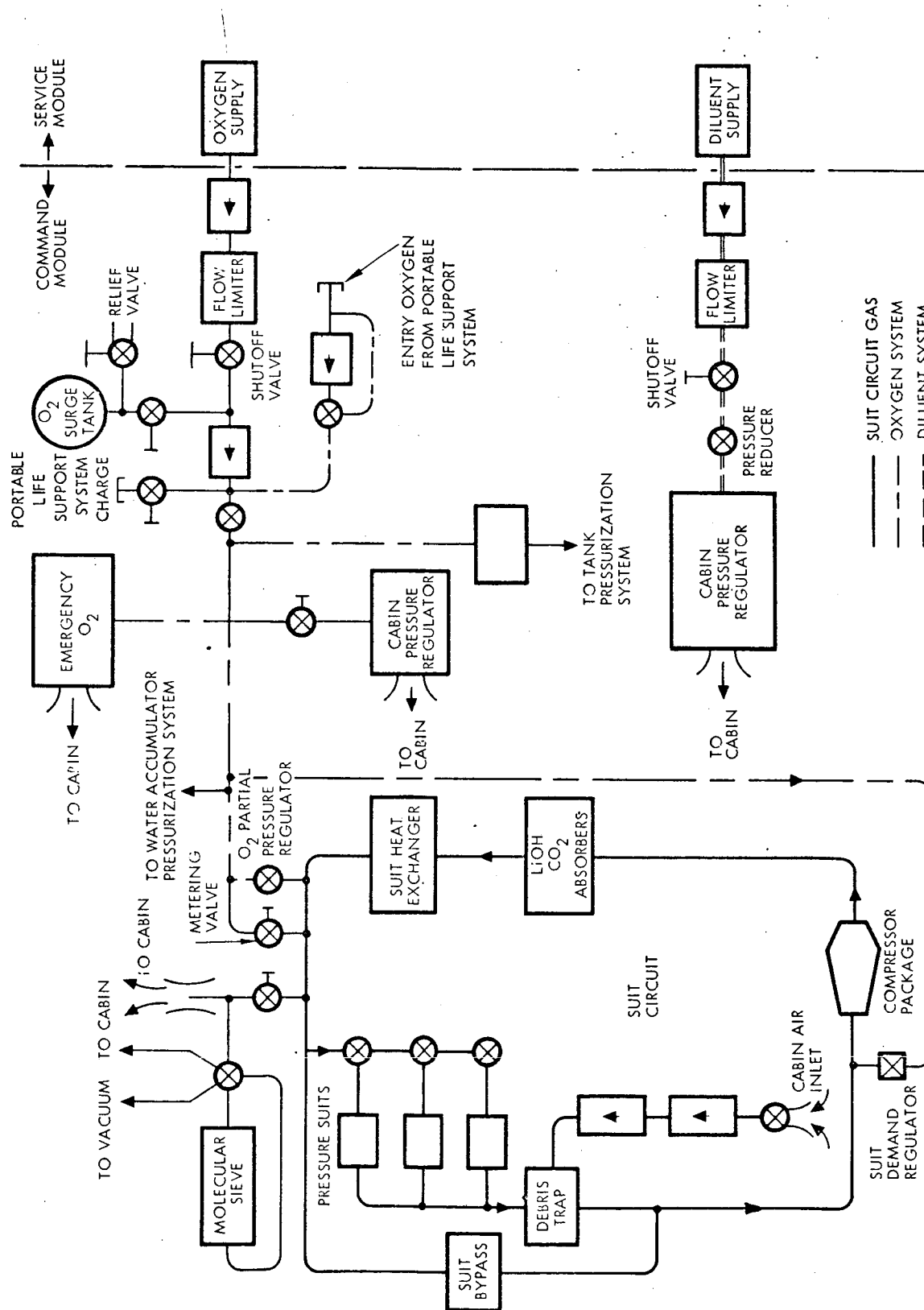
Figure 7. Atmosphere Supply and Control, Subsystem Block Diagram
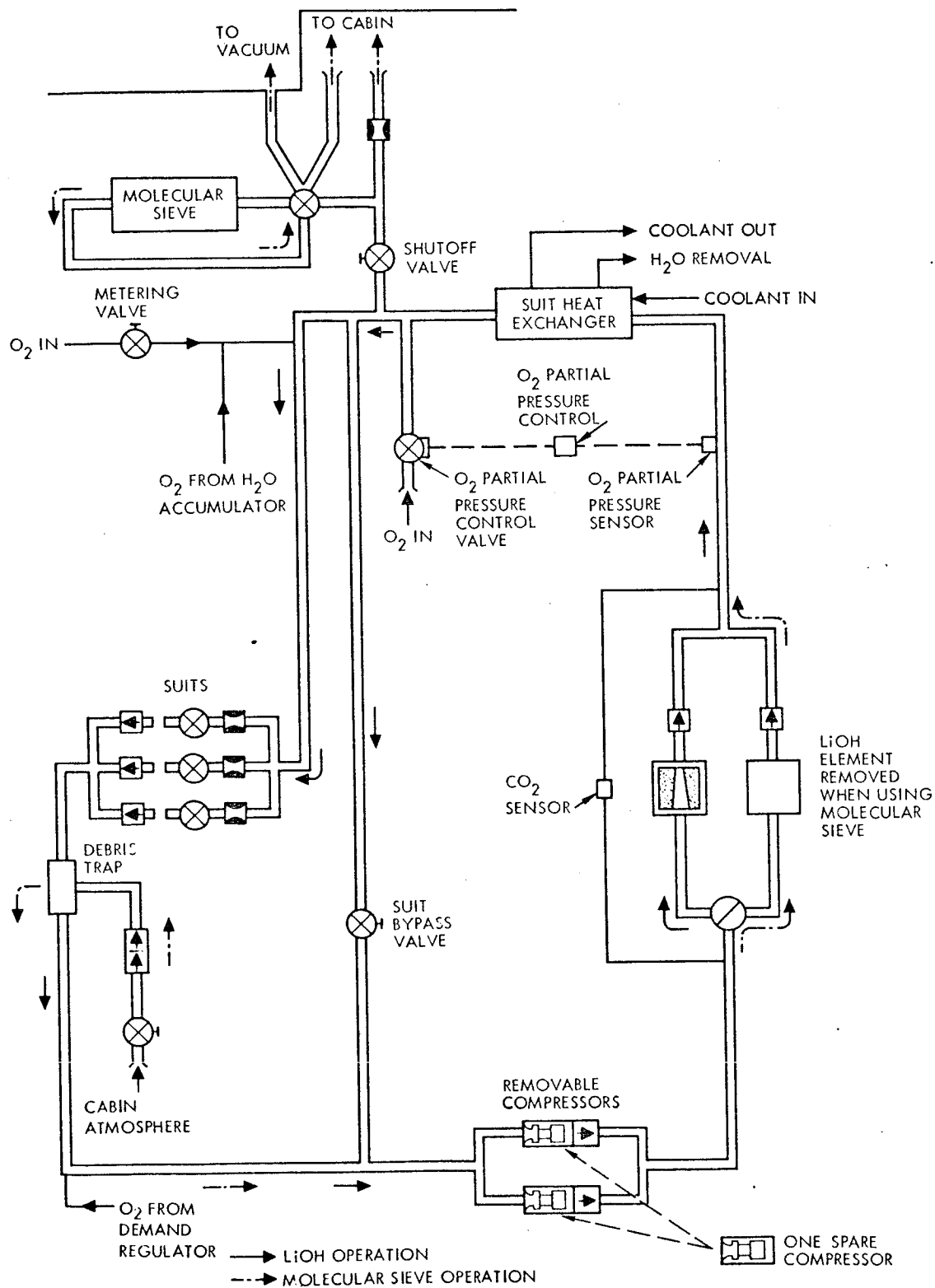
Figure 8.   Suit Circuit Schematic (Recommended Configuration)

will slowly decrease, as shown in Figure 9, for the change to an oxygen-nitrogen atmosphere. The diluent will flow to the cabin through the cabin pressure regulator in the diluent supply to maintain cabin pressure at 5.0 psia. The partial pressure control system will maintain the oxygen partial pressure at 3.5 psia. If the cabin is punctured, oxygen will flood the cabin through the emergency inflow control valve and the partial pressure valve. Some diluent will flow into the cabin through the cabin regulator, until the cabin pressure drops to 3.5 psia when the regulator shuts off automatically. It should be noted that the cabin pressure regulators for the oxygen and diluent systems are identical, and the capacities of this unit and the other control functions are sufficient for the AES requirements.
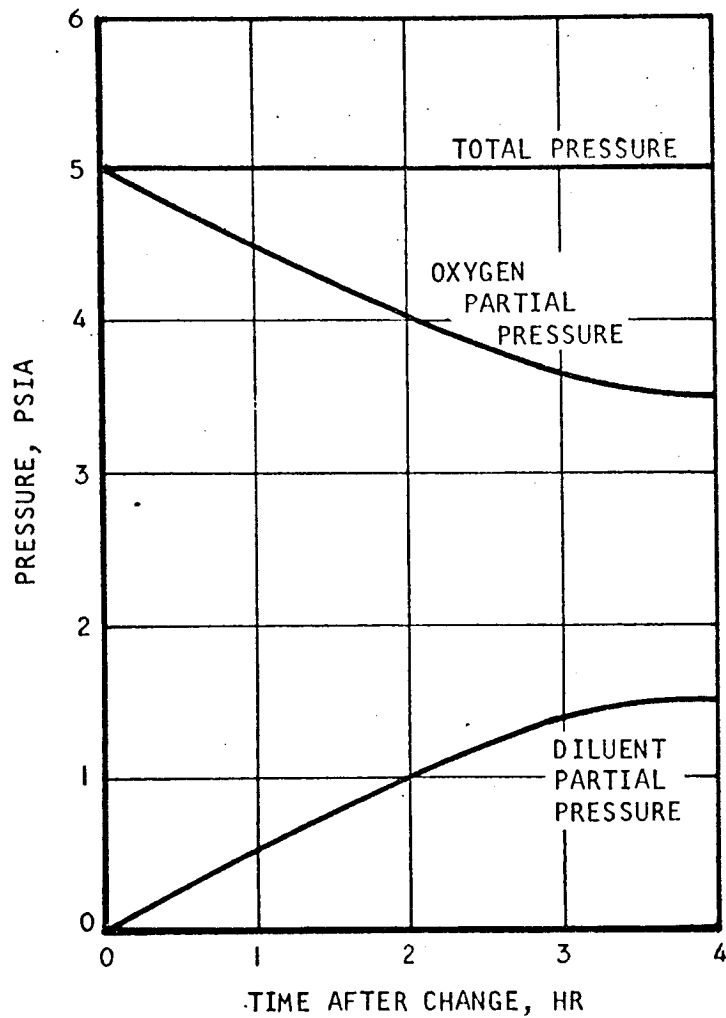
Figure 9.　Atmosphere Composition Response

CARBON DIOXIDE CONTROL

Several processes are competitive for providing carbon dioxide control for the AES mission. The AES general time schedule, however, limits the selection to processes and systems that have been developed already or that are in late development stages. The two processes for CO_2 control investigated in this study were nonregenerable absorption by lithium hydroxide (currently used in the Apollo CM) and regenerable adsorption using a molecular sieve. The first process is irreversible and characterized by the use of expendables, while the second is a cyclic process in which one bed is regenerated while the alternate is being used. Systems based on absorption by nonregenerable sorbents are in general very simple; the cyclic processes used in regenerable systems are by comparison more complex, but they offer weight savings for long-duration missions.

NONREGENERABLE SYSTEMS

The Apollo CO_2 control system uses expendable lithium hydroxide for absorption of CO_2 from the suit-loop gas circuit. The Apollo CO_2 removal canister consists of two parallel beds, with an integral latching and interlock arrangement permitting safe reloading of one bed in a vacuum environment. The canister is designed for CO_2 adsorption by lithium hydroxide at the rate of 0.29 pounds per hour. The weight of the empty canister is 19 pounds. A single lithium hydroxide charge weighs 4 pounds, and it is packed with 0.2 pounds of activated charcoal for odor removal. The charge is inside a plastic container that has a filter; the total expendable charge weight is 4.5 pounds. The CO_2 absorption capacity of one charge is 3.4 pounds (minimum), which corresponds to a utilization effectiveness of 92.5 percent. One charge is sufficient for control of CO_2 for a three-man crew for 12 hours. Thus, 68 charges would be required for the 34-day lunar mission, and 90 charges would be required for the 45-day earth orbital mission. This represents a total expendable charge weight of 306 pounds for the lunar mission and 405 pounds for the earth orbital mission.

Other nonregenerable absorption materials are candidates for CO_2 removal in the AES system. Their use in place of LiOH is ruled out on the basis that none of the approaches are developed to as high a degree and, therefore, a change is not warranted.

Table 30 compares the weight penalties of the nonregenerable systems considered during the study.

Table 30. Comparison of Nonregenerable Systems

Chemical	lb CO_2 Absorbed lb Compound	lb O_2 Generated lb Compound	Respiratory Quotient (RQ)	lb Compound Required Per Day*	Total Penalty (lb/day)**
LiOH	0.925	0		6.9	6.9
Li_2O	1.46	0		4.4	4.4
M_gO - AL_2O_3	0.26	0		24.5	24.5
Li_2O_2	0.925	0.36	1.87	6.9	3.68
KO_3	0.25	0.46	0.395	25.5	10.3
NaO_2	0.4	0.428	1.00	15.9	7.1

*Based on 2.12 lb CO_2 per man day
**Tankage penalty 1.3 lb/lb oxygen saved

The lithium peroxide (Li_2O_2) and sodium superoxide (NaO_2) appear to offer significant weight advantages over LiOH, when compared strictly on the basis of CO_2 removal capacity and oxygen generation. When these chemicals are integrated into a life support system other factors must be considered.

For example, the heat of reaction for sodium superoxide is 2150 Btu/lb of CO_2 removed which compares to 875 Btu/lb CO_2 for LiOH. The lower heat of reaction of LiOH is desirable since less thermal load is imposed upon the ECS due to CO_2 absorption.

To maintain reasonable values of heat of reaction, the dew point of the inlet process gas must be very low. Manned tests showed that predrying of process gas through silica gel was necessary to obtain the desired heat of reaction. For extended missions the silica gel must be regenerated either by exposure to vacuum or by thermal regeneration. These system complexities and additional weights are not included in the preceding table and if included would illustrate the weight advantage of LiOH.

The final disadvantage of all other candidate approaches is that they are not at the same stage of development as the man-rate LiOH. Therefore, LiOH is selected as the most applicable of the nonregenerable approaches considered here.

REGENERABLE SYSTEMS

Regenerable CO_2 removal systems based upon use of molecular sieve synthetic zeolite adsorbents and silica gel adsorbent have been investigated for a number of years for possible application to spacecraft atmosphere control systems. A number of engineering prototype systems have been built and operated for extended periods. Consequently, there is little doubt that a regenerable CO_2 removal system can be designed and developed for the AES missions, although no qualified flight hardware is available at present.

Molecular sieve adsorbents are not used by themselves to accomplish removal of CO_2 from a process gas stream because water vapor is adsorbed in preference to CO_2. Predrying is therefore necessary to preclude loading of the adsorbent with water vapor instead of CO_2. Predrying of the process gas by use of a condensing heat exchanger is generally unsatisfying since extremely low dew points on the order of -100 F are required and, therefore, predrying is usually accomplished using a sorbent bed such as silica gel.

The construction of a molecular sieve CO_2 removal system is characterized by a silica gel bed for predrying the process gas followed by a molecular sieve bed for CO_2 adsorption. The quantity of sorbents required for extended missions is reduced by using two such bed combinations, one bed adsorbing and the other bed being regenerated. The heat released as a result of adsorbing water vapor and CO_2 heats both the bed and the process gas stream. Likewise, when the bed is regenerated, the heat of desorption must come from the thermal capacity of the sorbent bed. This approach is called the adiabatic system. An alternate approach incorporates heat exchange surfaces within the two beds which are used to cool the adsorbing bed for more efficient CO_2 and water vapor removal and to heat the desorbing bed for more rapid regeneration. This approach is called the thermal swing system, where low and high temperature fluid is used to cool the adsorbing bed and to heat the desorbing bed respectively.

The desorbing bed for both the adiabatic and thermal swing systems is regenerated by exposing the bed to vacuum to accelerate the regeneration process. The CO_2 is, therefore, lost overboard as is the adsorbed water contained in the silica gel bed. The two bed adiabatic and the two bed thermal swing are therefore called vacuum dump systems.

Conservation of water adsorbed on the silica gel is possible for extended missions where a water surplus is not available onboard the vehicle. This approach includes placing the silica gel and molecular sieve adsorbents in separate canisters. The four canisters (two silica gel and two molecular sieve canisters) comprise the system in conjunction with gas and liquid valves. The process gas first passes through a silica gel bed, then into the

molecular sieve bed and finally through the second silica gel bed. The pass through the second silica gel bed regenerates this bed since the very dry process gas is capable of stripping water from the adsorbent. Heating of this bed is usually accomplished with hot fluid flowing through an internal heat exchanger to enhance regeneration. During this same time the second molecular sieve canister is heated by the hot fluid and the canister is vented to vacuum. The only materials dumped to vacuum are the CO_2 in the adsorbent and the volume of gas contained in the canister. When the first silica gel bed is loaded with water and the molecular sieve is loaded with CO_2, the flow path is changed such that the regenerated silica gel bed becomes the desiccant bed for the regenerated molecular sieve bed, and the loaded silica gel bed is regenerated by dry exhaust gas from the molecular sieve. The loaded molecular sieve bed is heated and exposed to vacuum for regeneration purposes.

The four bed system is usually designed as a thermal swing system because of the large heat effects associated with regeneration and adsorption of water in the silica gel (i.e., approximately 1300 Btu/lb). The CO_2 removal beds could be either thermal swing or adiabatic since the heat effects of adsorption of CO_2 are relatively small (i.e., 300 to 400 Btu/hr).

In all molecular sieve systems considered in the study, the molecular sieve CO_2 adsorbent is regenerated by exposure to vacuum. Whether the CO_2 adsorbent bed is operated under adiabatic or thermal swing conditions will significantly influence the capacity of the adsorbent, but will have relatively little effect on the basic system configuration. The method by which the water vapor is handled will be the most significant factor in determining basic system design. Three basic regenerable CO_2 removal systems were considered during the study: (1) two-bed adiabatic vacuum-dump system, (2) two-bed thermal-swing vacuum-dump system, and (3) four-bed thermal-swing system. The two vacuum-dump systems do not conserve water during desorption, and their application is limited to missions in which the fuel cell power system provides a surplus water supply. The four-bed system conserves the water during the purge-gas desorption from the silica gel adsorbent, and it is applicable to missions where surplus water is not available.

Two-Bed Adiabatic Vacuum-Dump System

A schematic of the two-bed adiabatic is shown in Figure 10. The process gas enters the CO_2 removal system through valve 1 and flows to the adsorbing bed through the center duct. The gas flows radially outward through the silica gel adsorbent, and then radially inward through the molecular sieve adsorbent. After passing through the adsorbing bed, the gas returns to the cabin through valve 2. The silica gel serves to predry the process gas to a dew point temperature lower than -50 F before the gas passes through the molecular sieve for CO_2 removal.
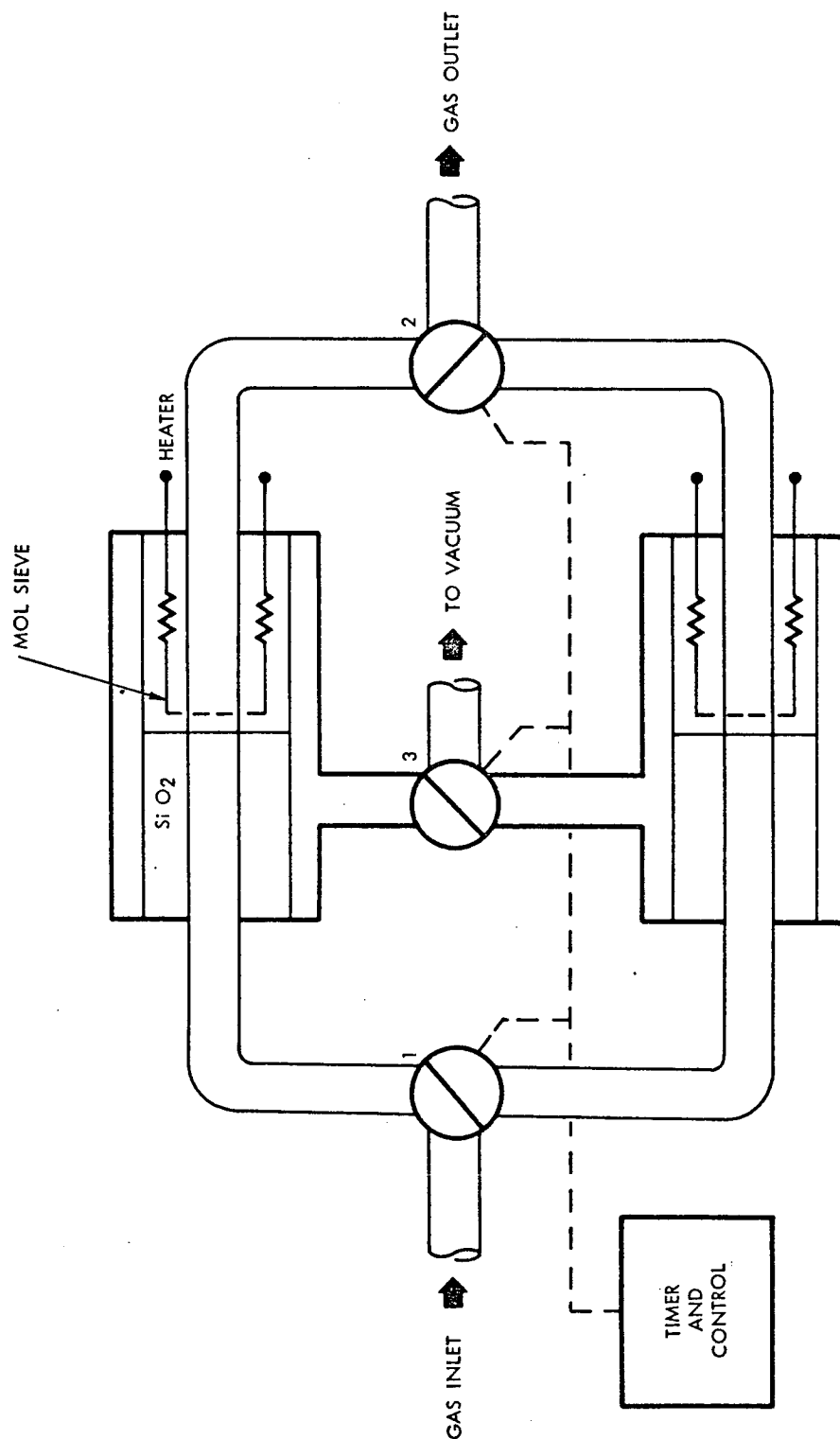
Figure 10.   Two-Bed Adiabatic Vacuum Dump

Valves 1 and 2 serve to close off the desorbing bed from the cabin atmosphere while valve 3 is opened to the space vacuum. The vacuum operates on the entire outer circumferential area of both the silica gel and molecular sieve adsorbents. The resulting flow during desorption is thus radially outward, which provides an efficient desorption method. Electric heaters located within the molecular sieve adsorbent are used periodically to remove the water contaminating the adsorbent.

Two-Bed Thermal-Swing Vacuum-Dump System

A schematic of the two-bed thermal-swing system is shown in Figure 11. The process gas enters the CO_2 removal system through valve 1, flows through the adsorbing bed, and is returned to ambient through valve 2. The two sorbent beds incorporate plate-and-fin heat exchangers. The adsorbent pellets are packed on the air side of the heat exchanger, with silica gel at the front end and molecular sieve at the back. With this type of bed construction, the adsorption and desorption temperatures can be controlled at approximately 60 F and 100 F, respectively.

Valves 1 and 2 also isolate the desorbing bed from the cabin atmosphere. The front end (water adsorption) of the desorbing bed is opened to space vacuum, while the back end (CO_2 adsorption) of this bed is isolated by valve 2. This single-end vacuum-desorption design is based on the results of tests conducted in the development of this system. Tests were made of other desorption schemes, such as double-end desorption and center-point desorption, but neither of these approaches provided as satisfactory performance as the single-end desorption.

The 55 F coolant enters the system at valve 3, and the 125 F hot coolant enters the system at valve 5. Valve 3 directs the cold coolant to the adsorbing bed, and valve 4 directs the hot coolant to the desorbing bed. After passing through the heat exchanger in the beds, the cold coolant is directed out of the system by valve 7, and the hot coolant is directed out of the system by valve 6.

Four-Bed Thermal-Swing System

A schematic of the four-bed system is shown in Figure 12. The silica gel desiccant and the molecular sieve adsorbent are contained in separate canisters. These canisters are of the same plate-and-fin heat exchanger design as that used in the two-bed isothermal system. The process gas flows through the adsorbing silica gel and molecular sieve beds as controlled by valves 1 and 2. At the outlet of the adsorbing molecular sieve bed, the gas is essentially free of CO_2 and has a dew-point temperature lower than -100 F, its temperature is about 50 F. This gas is then directed to the desorbing silica gel bed by valves 3 and 4, where it is used as a purge gas to
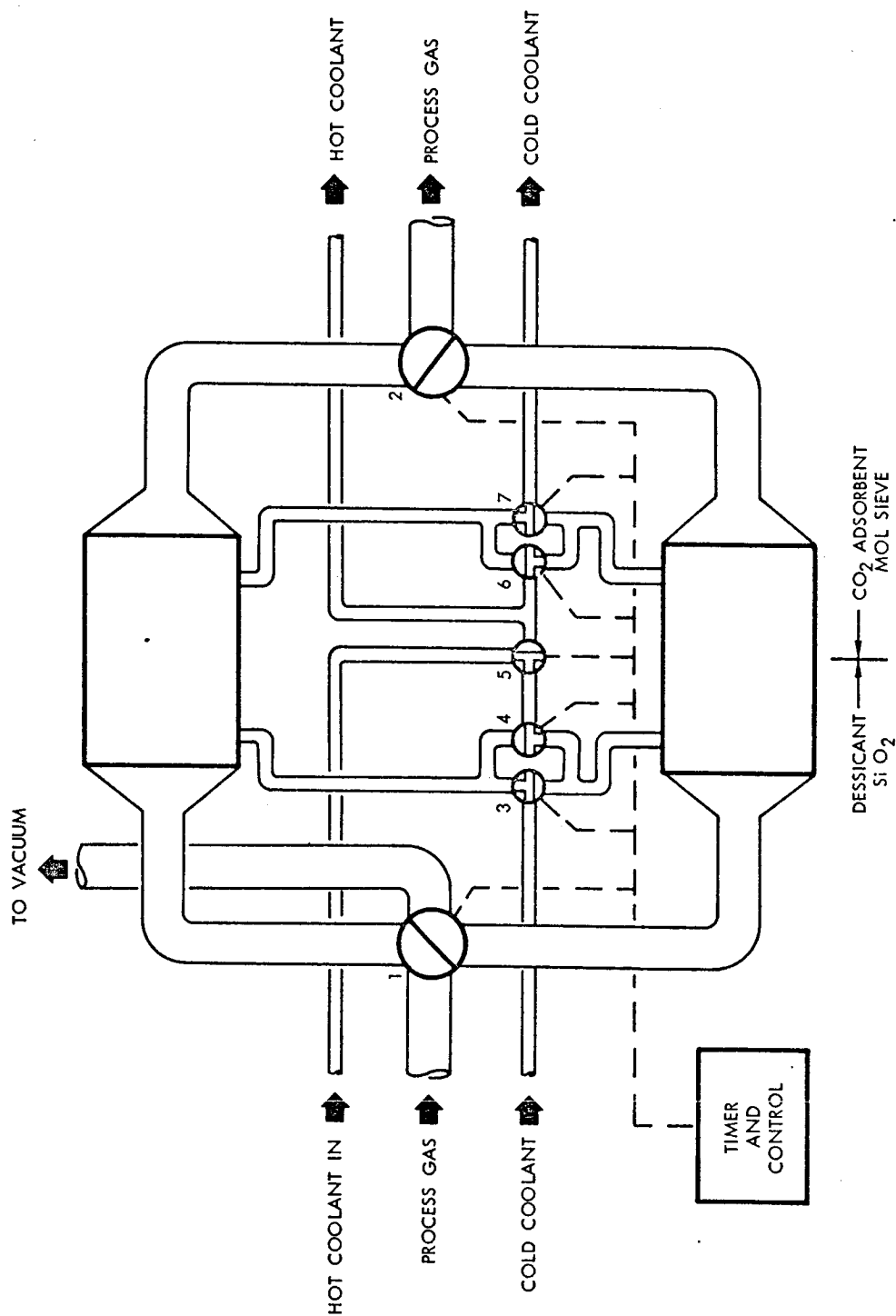
Figure 11.   Two-Bed Thermal Swing Vacuum Dump

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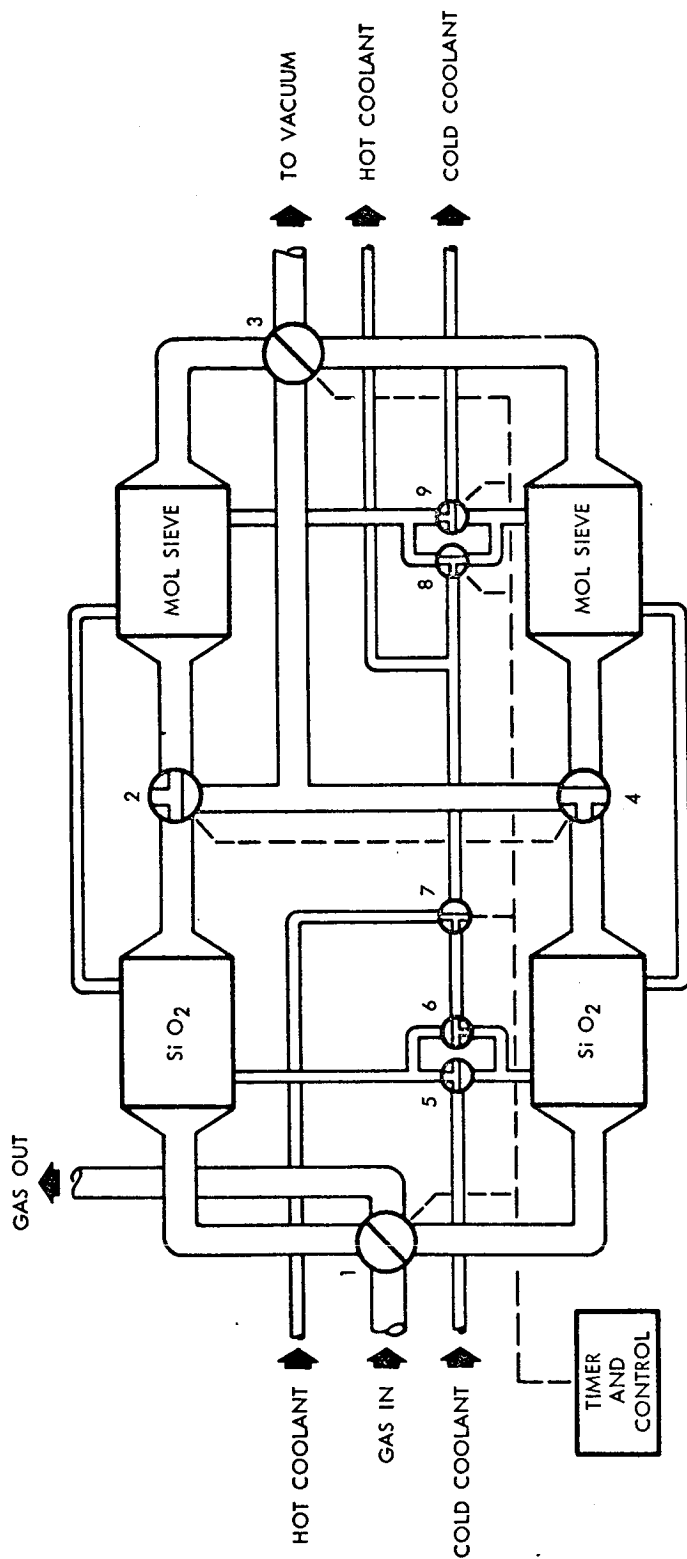
Figure 12. Four-Bed Thermal Swing System

desorb moisture from this loaded bed. Heat is transferred to the desorbing silica gel partly by conduction from the heat exchanger surface and mainly by means of the purge gas. In this hot purge-gas desorption method, the heat transfer mechanism is by forced convection between the hot heat exchanger surface, the gas, and the solid adsorbents, resulting in very efficient heat transfer. Desorption of the molecular sieve is by vacuum, with hot coolant circulating through the heat exchanger. Valve 3 opens the desorbing bed to space vacuum, while valve 4 isolates the bed from the cabin atmosphere.

CO_2 REMOVAL SYSTEMS TRADE-OFF

Table 31 summarizes the results of the preliminary trade-off studies performed and indicates the performance of the respective systems. The system comparison is based on a reliability for mission success greater than 0.999 for all systems. A power weight penalty basis of 1.7 pounds per kwhr for the 45-day mission was used, and the radiator penalty was taken as 0.01 pounds per Btu per hour. The power requirements were based on the flow rate and pressure drop of the CO_2 system only.

Figure 13 shows a comparison of the weight of the various CO_2 removal systems on the basis of mission duration. The two-bed and four-bed thermal-swing systems show a small increase in weight with mission time due to the power and gas loss penalty. The regenerable sorbent systems show dependence upon mission time due to the power consumption and gas loss. The adiabatic system results in approximately 50 percent greater weight penalty than the two-bed thermal-swing system, and is dependent upon mission time, primarily because of the accumulative gas loss penalty caused by dumping the large-volume canister. The use of many LiOH charges in the current Apollo Block II configuration results in the greatest weight and is strictly time-dependent, since the approach is based upon expendable supplies.

Figure 14 illustrates the volume penalty associated with carbon dioxide control. Included in the total volume penalty, besides the physical size of the system, is the volume of a two-day LiOH supply for closed suit operation, as well as the volume associated with storage of oxygen and hydrogen for power, and oxygen and nitrogen to make up for gas losses. More than 10 cubic feet of space can be saved by use of a regenerable carbon dioxide control system.

Both the weight and volume penalty relationships indicate that the two-bed thermal-swing system offers advantages of lower weight and volume penalty when compared with the four-bed system or the two-bed system. All regenerable systems are far superior to that involving the use of expendable LiOH for the range of mission durations studied.

Table 31. System Comparison

Mission	LiOH		Two-Bed Adiabatic Vacuum-Dump		Two-Bed Thermal-Swing Vacuum-Dump		Four-Bed Thermal-Swing	
	34-day	45-day	34-day	45-day	34-day	45-day	34-day	45-day
Cycle time (min)			15	25	25	25	25	25
CO_2 dynamic removal efficiency			0.95	0.95	0.95	0.95	0.95	0.95
Pressure drop, in. H_2O	2	2	1	1	1.3	1.3	3.3	3.3
Equivalent power required, W (ac)	20	20	10	10	13	13	33	33
Maximum cooling load, BTU per hour	0	0	0	0	530	530	530	530
Weight penalty, lb								
Fixed	306	405	75	91	34	34	48	48
Two-day LiOH supply	0	0	18	18	18	18	18	18
Gas loss (7 psia)	0	0	44	54	6.9	8.3	4.8	5.7
Power [1]	32.5	42.5	19.3	25.5	25.1	33.2	63.2	83.6
Radiator [2]	0	0	0	0	5.3	5.3	5.3	5.3
Total Weight Penalty, lb	338.5	447.5	156.3	188.5	89.3	98.8	139.3	160.6
Volume penalty, cu ft								
Two-day LiOH supply	0	0	0.64	0.64	0.64	0.64	0.64	0.64
System	10.8	14.4	0.96	1.2	0.30	0.30	0.62	0.62
Power and gas loss	0.8	1.1	1.04	1.34	0.60	0.80	1.48	1.94
Total Volume Penalty, cu ft	11.6	15.5	2.64	3.18	1.54	1.74	2.74	3.20

[1] Power penalty, 1.7 lb/KWH for 45-day mission

[2] Radiator penalty, 0.01 lb/Btu/hr
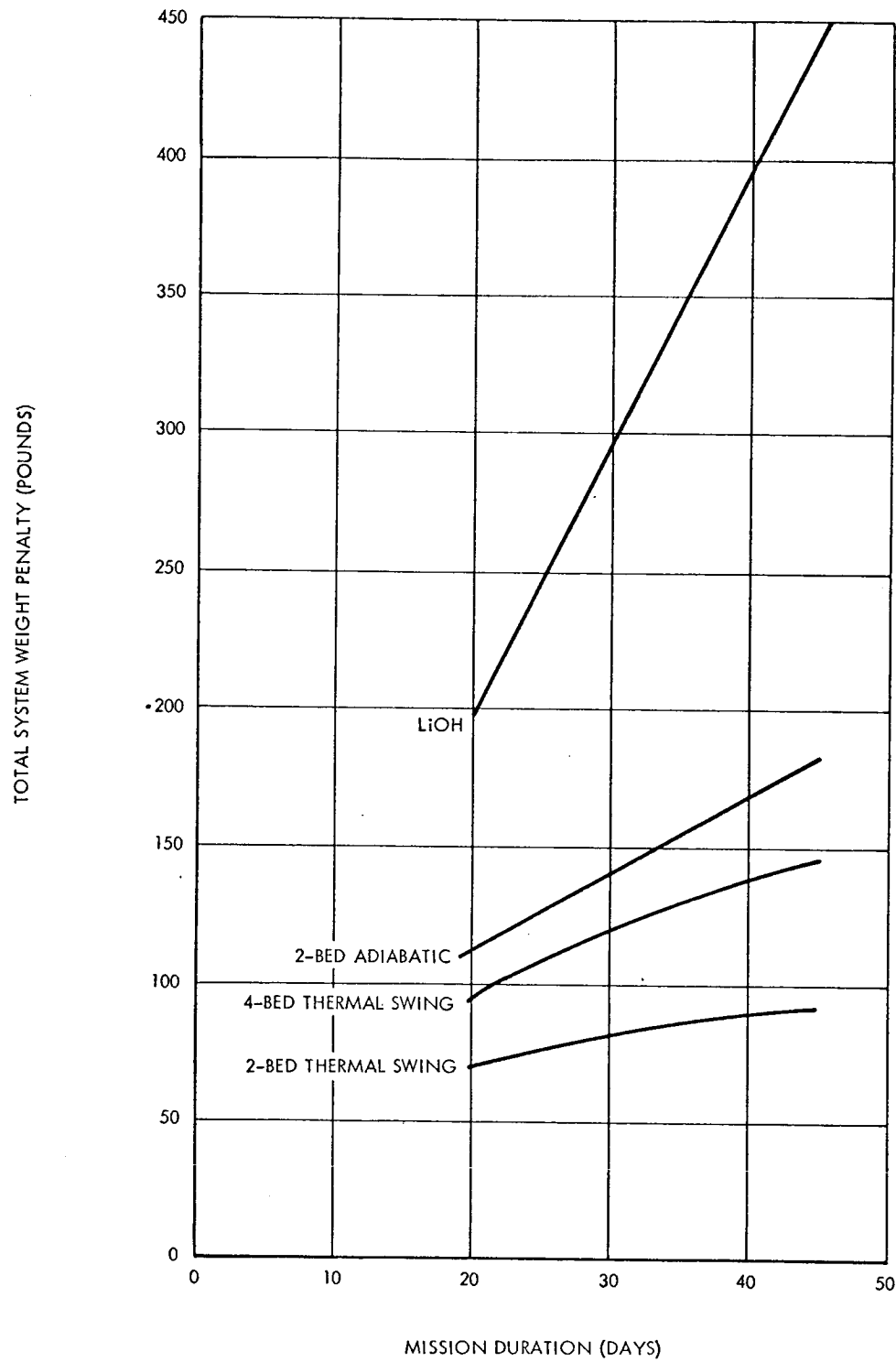
Figure 13. LiOH/Molecular Sieve Weight Penalty Comparison
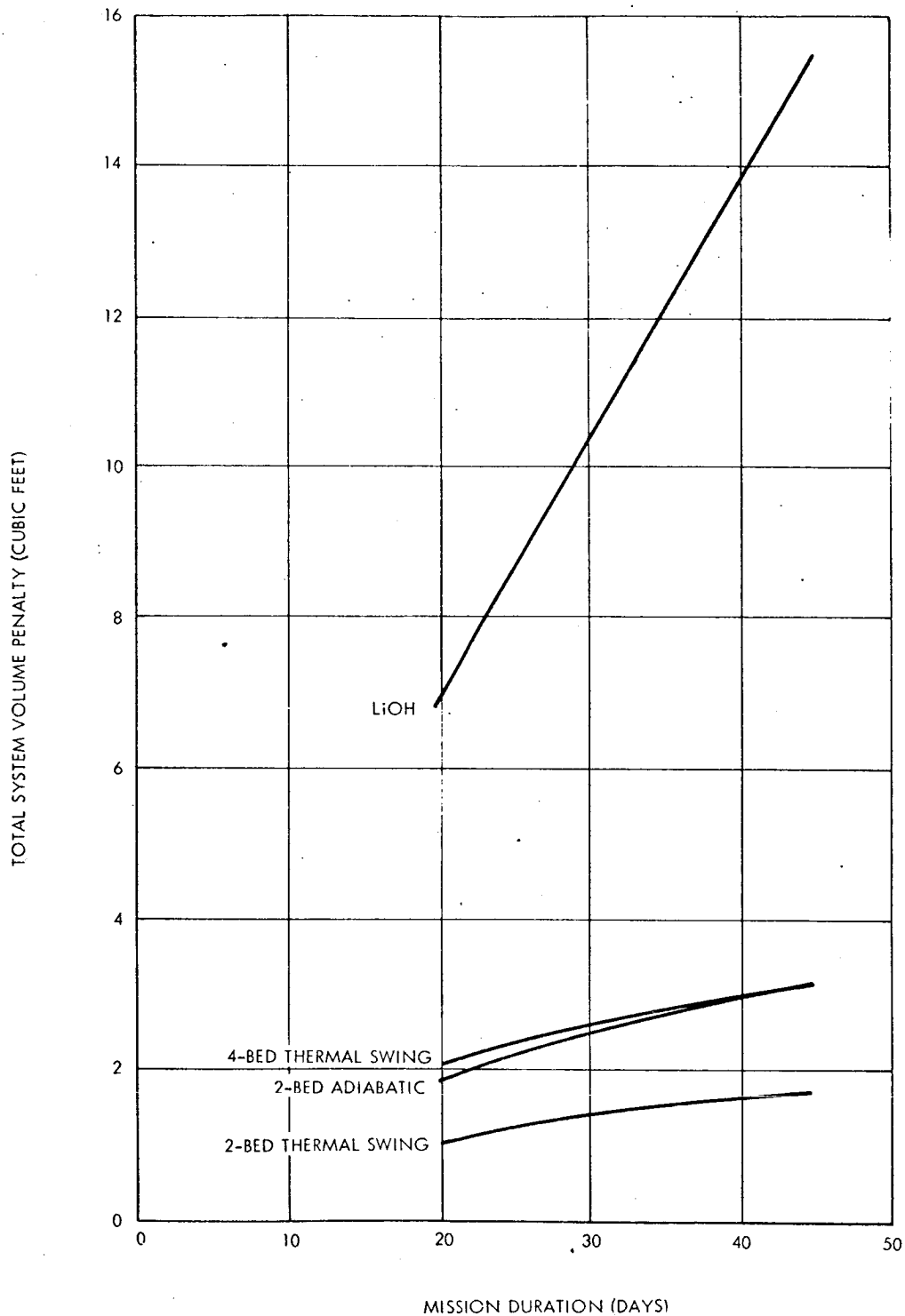
Figure 14.   LiOH/Molecular Sieve Volume Penalty Comparison

It should be noted that the expendable weight of the LiOH charges required for shirtsleeve operation are included in the molecular sieve system weight penalty. It was assumed that two man-days of closed suit operation would be required. Thus, 18 pounds of lithium hydroxide is chargeable to the molecular sieve system.

The total system weight penalties for the 34-day lunar and 45-day earth orbital missions are summarized in Table 32.

Table 32. Total CO_2 System Weight Penalty

System	34-Day Mission	45-Day Mission
LiOH	338.5	447.5
Two-bed adiabatic	153.5	184.5
Two-bed thermal-swing	85.2	93.1
Four-bed thermal-swing	129.6	147.0

Utilization of a two-bed thermal-swing system for the 45-day mission results in a weight saving of 354.4 pounds over a system using LiOH, while the four-bed system saves 300.5 pounds. Possibly equally as important, more than 10 cubic feet of space is saved by the use of a molecular sieve system. Thus, on the basis of weight and volume, a molecular sieve system is preferable for the AES mission. The decision to use a two-bed or four-bed thermal-swing system must be made on the basis of the vehicle water balance. As stated previously, an adiabatic system is not recommended because of performance deterioration and weight and volume penalties. The water balance in succeeding sections shows that the two-bed system is favored.

The systems trade-off data shown in Tables 31 and 32 and in Figure 13 are the results of preliminary investigation, and do not include the penalties for integration of the molecular sieve into the existing ECS. The thermal-swing type molecular sieve requires hot and cold liquid for desorption and adsorption; these liquids logically should be obtained from the existing water-glycol coolant system. Providing the piping connections to the sieve from the coolant system is not an insurmountable problem, but obtaining coolant at the proper temperature levels for efficient sieve operation is a problem of somewhat greater magnitude.

AES, TWO-BED THERMAL-SWING MOLECULAR SIEVE SYSTEM DEFINITION

The preceding discussion established the two-bed thermal-swing molecular sieve system as the optimum CO_2 removal system. The following discussion presents the optimum design configuration of the two-bed thermal-swing molecular sieve system for the AES mission.

1. CO_2 production rate　　　　　　　　　　0.265 lb/hr

2. Total atmosphere pressure　　　　　　　5.0 psia

    a.　3.5 psia O_2 partial pressure

    b.　Diluent, as required to maintain 5 psia total pressure

3. System inlet CO_2 partial pressure　　7.0 mm Hg

4. System inlet H_2O partial pressure　·　10 mm Hg (52 F dew pt)

5. Cold coolant temperature (maximum)　58 F

6. Hot coolant temperature　　　　　　　125 F

7. Cold coolant flow rate　　　　　　　　167 lb/hr

8. Hot coolant flow rate (minimum)　　　25 lb/hr

9. System CO_2 removal efficiency　　　　95 percent (from parametric analysis)

10. Mission bed depth　　　　　　　　　　6 in. (from parametric analysis)

The CO_2 production rate selected for design is based on the average metabolic activity level of the crew. The design CO_2 partial pressure was derived from atmosphere interchange considerations, assuming that the CO_2 removal unit is located in the CM and that the CO_2 partial pressure in the laboratory module is 7.6 mm Hg with average CO_2 production rate from three men. It is anticipated that under these conditions the CO_2 partial pressure could exceed 7.6 mm Hg for a short time (10 to 15 minutes) during the periods of high metabolic rate corresponding to the exercise function of

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one crew member. The inlet water vapor partial pressure was obtained from analysis of the transient humidity level in the cabin during the exercise period of one of the crew members. The cold coolant temperature is representative of the maximum value anticipated in the CM thermal loop; the hot coolant flow rate is the minimum anticipated. These extreme conditions were selected for system design to preclude system performance degradation by water poisoning of the molecular sieve such as would occur if the system were designed for nominal conditions.

The bed depth is limited to 6 inches to ensure effective desorption of the sorbents consistent with the single-end vacuum-desorption approach. The sorbent canister selected is a cross flow, plate-fin heat exchanger fabricated in stainless steel with nickel fins on the gas side; the fin characteristics of this unit are as follows:

Process gas side fins: 0.3 in. high, 4 fins/in. 0.004-in. nickel

Coolant side fins: 0.05 in. high, 20 fins/in. 0.10-in. offset, 0.002-in. stainless steel

The sorbents are packed between the gas side fins. The particular gas fin geometry selected yields good sorbent packing characteristics and heat transfer to the sorbents during adsorption and desorption. Stainless steel and nickel construction is selected based on high structural integrity, reliability for the service intended (vacuum exposure), and high resistance to corrosion.

A gas passage must be provided to dump CO_2 and water vapor, accumulated during the adsorption cycle, into space during the desorption cycle. This vacuum dump gas passage starts at the back ends of at least two molecular sieve units which are manifolded into a single vacuum dump line to carry desorbed gases to space. The vacuum dump line will make one penetration through the pressure shell of the CM and the gas passage must be continued from that point to space vacuum.

Detailed analysis of the Apollo Block II cold plate network configuration revealed that it is impossible to supply the molecular sieve with 125 F from the outlet of the unmodified Block II cold plate network. Several rearrangements of the Block II network were developed using AES heat loads to determine the feasibility of obtaining 125 F coolant fluid at a sufficient flow rate to meet the molecular sieve requirements. The coolant flow rate required during the desorption cycle is estimated at between 20 to 30 pounds per hour. Again, a higher coolant fluid flow rate is desirable to improve the performance of the adsorbent bed regeneration. One rearrangement of the Block II cold plate network shows it is possible to produce 21 pounds per hour

of coolant fluid at 125 F. However, the cold plate rearrangement developed would not meet the Block II cold plate design specification. More specifically, the base temperature of several of the cold plates would be higher than the allowable base temperatures. Therefore, it appears that an electrical heater would be a definite new requirement to supply the molecular sieve with the hot coolant fluid unless some of the cold plates whose allowable base temperatures were exceeded as a result of the Block II network rearrangement are redesigned to accept higher base temperatures.

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ECS WATER MANAGEMENT ANALYSIS

The water management system encompasses the sources of water, the distribution system including sensors and controls, the storage components, the heaters and coolers, and the components where the water is finally consumed. The water available is classified in two categories: (1) potable water, which may be used for astronaut consumption as well as all other uses; and (2) waste water, which may be used in any, except for astronaut consumption.

The only sources of water in the spacecraft during the missions are the fuel cells, which produce about 0.77 pound of water per kilowatt-hour of electricity generated, and the water which is loaded prior to launch in either the potable or waste water tanks. Because only a limited amount of water may be loaded before launch, the majority of water for the mission must be produced by the fuel cells. Although it is not a true source, some water is recovered in the suit heat exchanger as the moisture in the air condenses when the air is cooled. Similarly, some water is also recovered under conditions when LiOH is used for removal of CO_2 from the cabin atmosphere. All recovered water is treated as waste, and cannot enter the potable water system.

LUNAR ORBITAL MISSIONS

Lunar orbit abort requirements present the major consideration in the water management investigation. Sufficient water must be stored in the vehicle to allow safe return to earth under the abort criteria. Because of the time period involved (approximately 108 hours), abort from lunar orbit requires much more water than any other abort condition. Lunar orbit abort requirements were estimated as follows:

Assumptions

1. Two fuel cell continuous minimum load

2. Estimated 108 hours abort return

3. Three astronauts in their suits (suit heat exchanger is not on the redundant loop).

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Water Production

        1126 watts x 0.77 lb/kwh x 108 hr = 93.5 pound

Water Usage

      0.416 lb/hr astronaut consumption

   -0.1 lb/hr recovery in CO_2 removal by LiOH

     0.316 lb/hr

 +1.395 lb/hr suit heat exchanger requirement

    1.711 lb/hr x 108 hr = 184.5 pound

Suit heat exchanger	184.5 lb
Reentry	+ 7.0 lb
Contingency	+ 2.0 lb
	193.5 lb usage
	- 93.5 lb production
Storage requirement for lunar orbit abort	100.0 lb

    Figure 15 presents the water production and water consumption rates for the 20-day LEM escort reference mission. Until the time of lunar orbit insertion there is a large excess of water; however, the stringent storage requirement for abort causes a shortage of 22 pounds of water unless 22 pounds of water is added before launch. After LOI, an excess of water is produced in all phases of the mission.

    Figure 16 presents the water production and usage rates for the 34-day lunar polar orbit reference mission. The curves are almost identical to those shown in Figure 15 with exception of the lunar orbit phase between mission times of 75 and 750 hours. Again there is an excess of water until the time of LOI, when the abort requirement establishes a shortage of 22 pounds, which may be loaded on-board the vehicle before launch. After LOI, water production is in excess of usage.
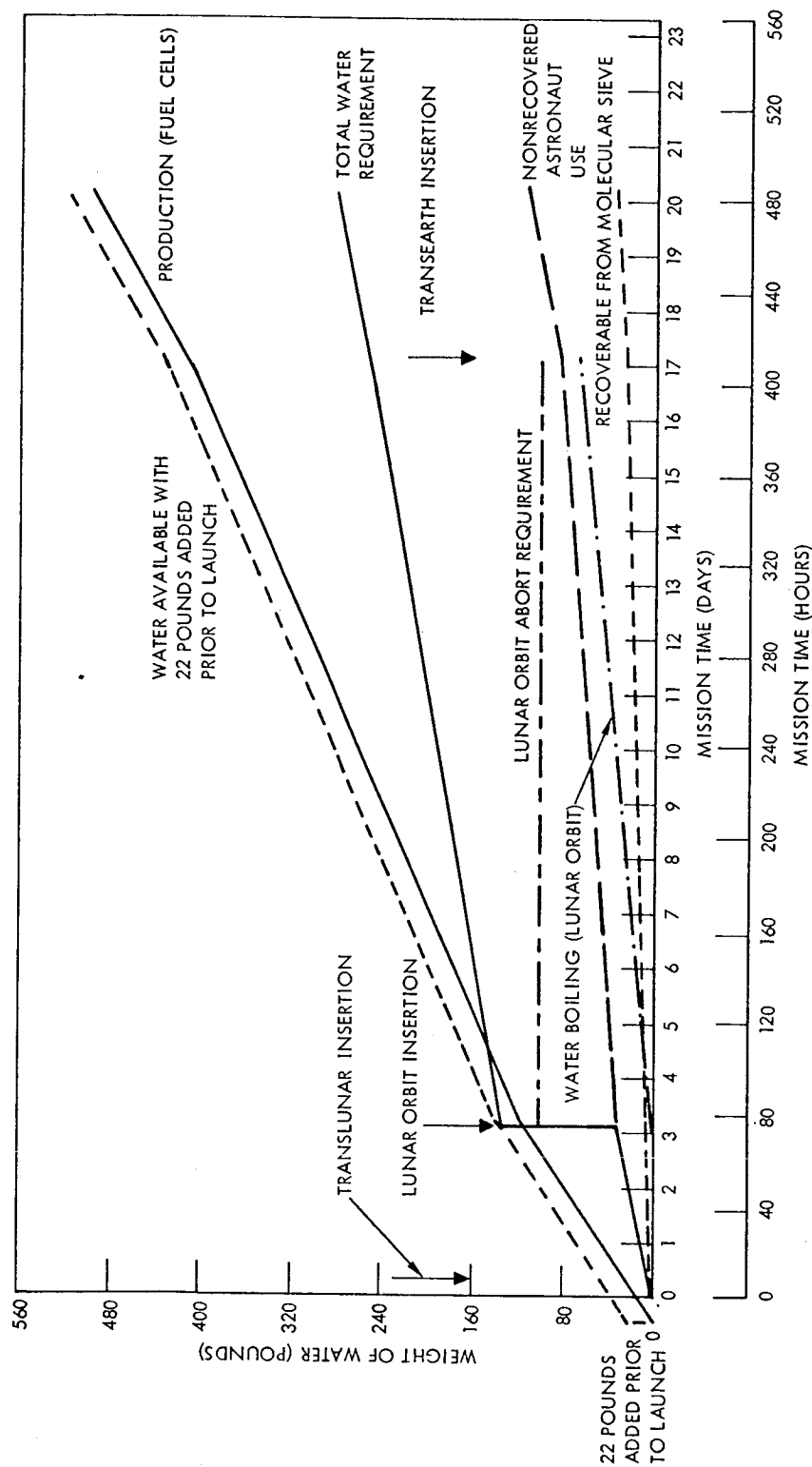
Figure 15.   Water Production and Usage, LEM Escort, Reference Mission 4

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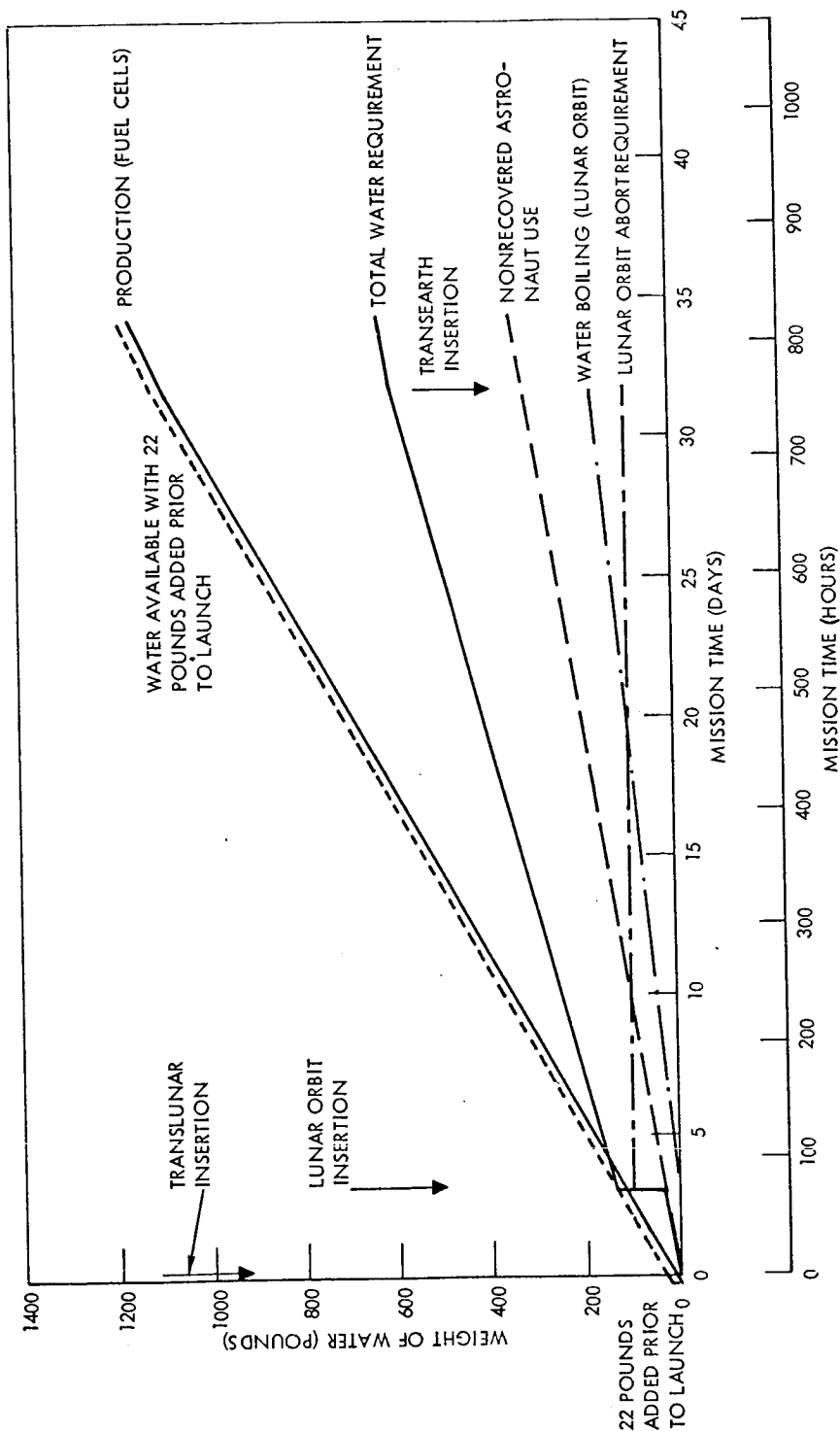
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Figure 16.   Water Production and Usage, Lunar Polar Orbit, Reference Mission 3

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EARTH ORBITAL MISSIONS

The earth orbital missions do not have as severe water storage requirements for abort conditions as lunar missions. The maximum abort time for earth orbit is about 6 hours compared to the 108 hours for lunar orbit. Consequently only 14.1 pounds of water are required for abort: 7 pounds for entry, 2 pounds for peak heat load boiling requirements, and 10.3 pounds for suit heat exchanger water boiling with 5.2 pounds produced by the fuel cells.

Figure 17 illustrates the water management situation in earth polar orbit. In every phase of the mission, water production exceeds water usage and there is no requirement to load any water before launch.

The earth synchronous orbital missions can be treated as cislunar cases as far as external environmental conditions are concerned. As such, some type of passive temperature control (PTC) will be used and water boiling will be required for only a small portion of the heat rejection. The water boiler will be used only for very short time peak heat loads (usually less than one hour in length) which occur only a few times in the duration of each mission. The amount of water boiled off in each one of these peak heat loads is usually less than one pound; the total for the entire mission is insignificant when compared to the total amount of water produced. The water management situation for this mission is shown in Figure 18.

WATER MANAGEMENT CONCLUSIONS

The conclusions of the water management analysis are as follows:

1. Mission abort requirements in lunar orbit are the controlling factors in the worst-case water management analysis of the four AES reference missions. At the time of lunar orbit insertion approximately 100 pounds of water must be stored to satisfy the requirements of the men and equipment for return to earth.

2. The 20-day LEM escort mission presents the most stringent water management requirements because of the water boiling requirements in lunar orbit and the amount of water stored to satisfy lunar orbit abort conditions.

3. In order of decreasing water requirements for mission abort, the missions are: (1) LEM escort (low inclination lunar orbit), (2) lunar polar orbit, (3) earth synchronous orbit, and (4) earth polar orbit. The first two missions have maximum abort times
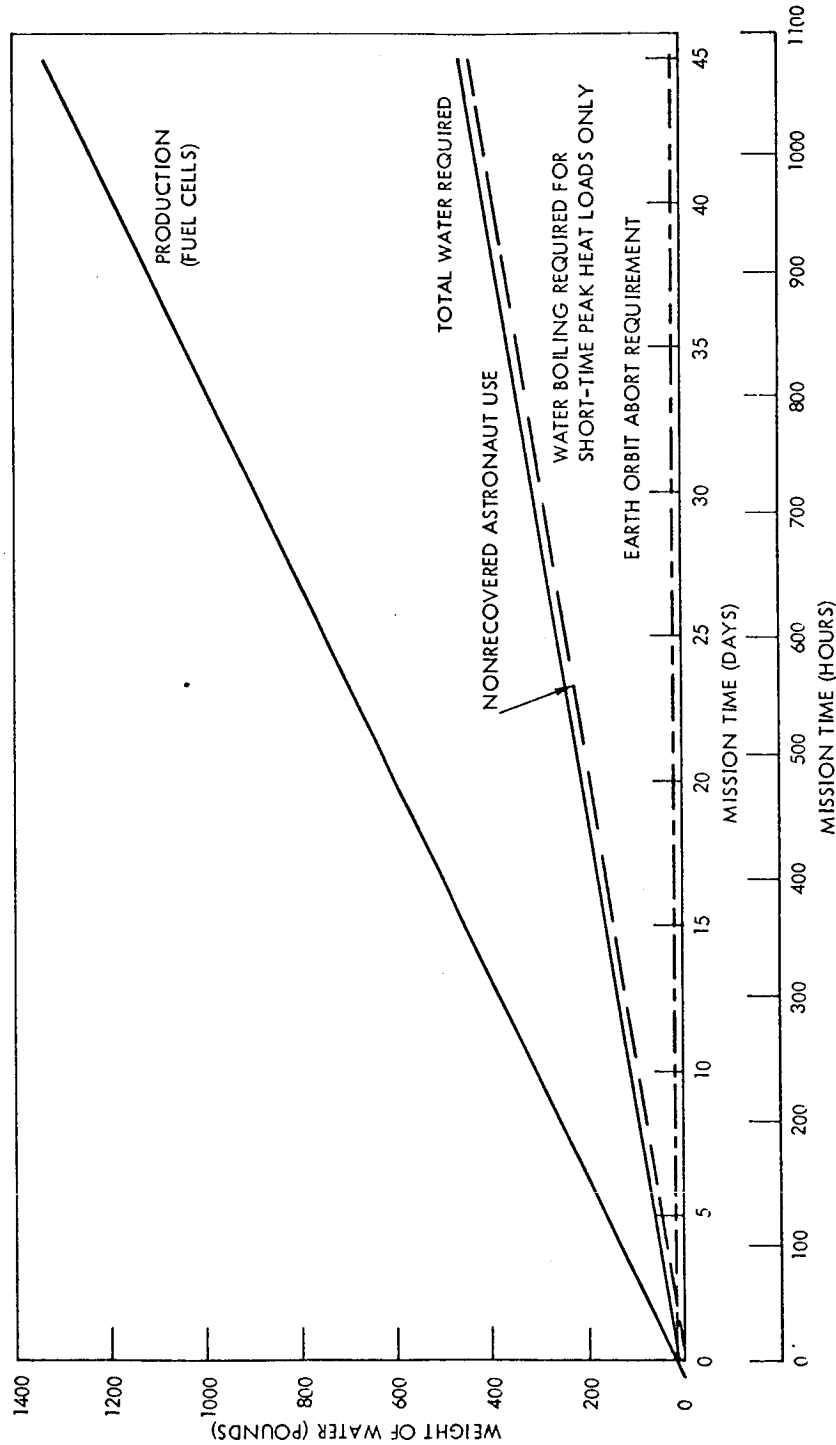
Figure 17.   Water Production and Usage, Earth Polar Orbit, Reference Mission 1

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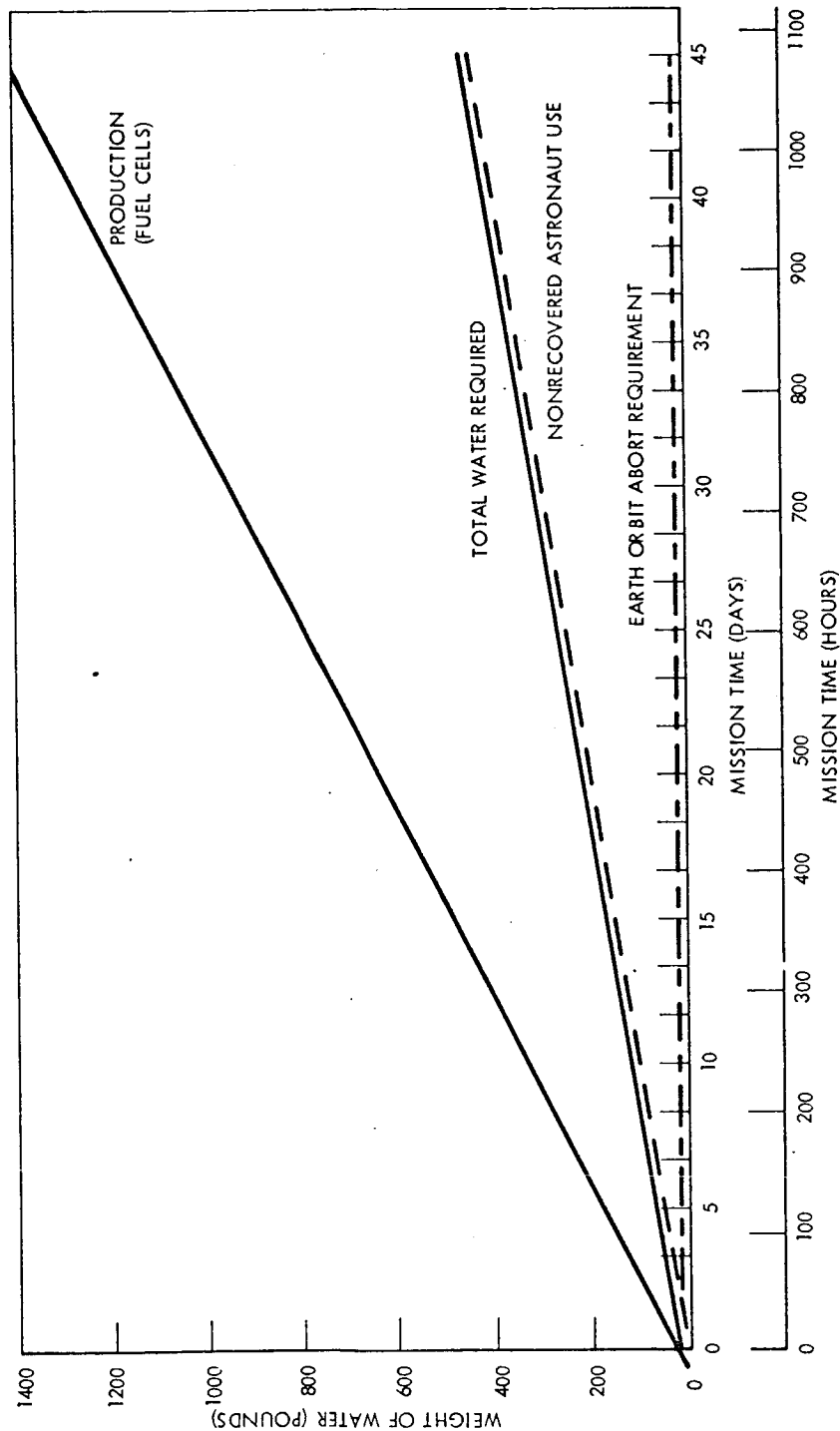
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Figure 18. Water Production and Usage, Earth Synchronous Orbit, Reference Mission 2

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of about 108 hours, requiring large amounts of water, while the last two missions have abort times of less than 6 hours, with little effect on water management considerations.

4.   To satisfy lunar orbit abort requirements an additional 22 pounds of water must be on-loaded prior to launch. The volume of the water tanks must be increased to allow sufficient water to be stored for lunar orbit abort.

ATMOSPHERE INTERCHANGE

Studies were made to determine the effect of compartment atmosphere interchange on the control of carbon dioxide, humidity, oxygen concentration, and temperature. The tabulation below presents the minimum atmospheric-flow requirements for each of the parameters with three men in the laboratory module (LM):

Parameter	Minimum Atmospheric Flow (cfm)
Humidity	120
CO_2	60
Temperature	425
O_2	40

The atmospheric flow rate for humidity control is based upon a dew-point differential of 3 F between the two compartments. The oxygen flow rate is based upon a 2-mm Hg difference in oxygen partial pressure, and the interchange rate for carbon dioxide is based upon a carbon dioxide partial pressure difference of 0.5 mm Hg and nominal generation of carbon dioxide. The use of simple compartment atmosphere interchange for thermal control of both compartments is based upon the temperature limits of 75 F ±5. The LM can then be at 80 F while the CM can be at 70 F. The 10 F temperature difference imposes a severe penalty in terms of fan-power to accomplish all the thermal conditioning by means of this approach. From the tabulation, it is clear than an interchange flow rate of 120 cfm is required to maintain a 3 F dew-point differential. At this flow, carbon dioxide and oxygen partial pressure control requirements are amply satisfied.

The fan selected for intercompartment circulation is the CM post-landing ventilation fan without modification. The characteristics of this fan are listed below:

Flow	180 cfm
Pressure rise	0.2 in. H_2O
Weight	4.5 lb
Power	13 watts

The use of this fan is recommended because of the relatively low power requirement associated with the high flow rate. Although the flow rate is 50 percent higher than necessary to control the dew-point in the two compartments to less than a 3 F differential, the extra circulation is recommended because the atmosphere differences between compartments will be less and the flow distribution within the LM will be more uniform.

The fan circulates the CM atmosphere into the LM by use of a 5-inch diameter duct located in the CM/LM tunnel. The duct is a rigid tube fastened to the side of the tunnel after mating pressurization of both modules. The exit of the duct located in the LM exhausts the circulated gas along the wall of the LM to avoid short-circuiting the gas flow. The tangential injection also enhances the mixing and purging of the LM atmosphere.

ELECTRONIC EQUIPMENT COOLING ANALYSIS

The major portion of the electronic equipment is cooled by a cold plate system. This system utilizes a liquid cooled cold plate on which the electronic package is mounted rigidly with a thermal grease applied between the package base and the cold plate which provides good thermal contact between the package base and the cold plate. This is especially important when the cabin is depressurized depleting the air which serves as a thermal conduction path. Basic AES cold-plate design is identical to Block II.

Nominal coolant flow rate through the cold plate system is 200 lb/hr. This flow is distributed throughout the network in accordance with the cooling needs of each electronic package, with an effort to keep the system pressure drop to a practical minimum.

The electronic cooling (cold plate) system was evaluated with respect to: (1) equipment heat loads based on timelines for AES missions, and (2) effects of additional cold plates and provision for temperature controlled coolant to the molecular sieve. Maximum heat loads for each electronic component influence the design of each individual cold plate whereas, the timeline heat loads for the complete cold plate system influence other ECS subsystems such as cabin heating, radiator performance and molecular sieve heating. The Block II cold plate network was evaluated considering several changes in order to provide high temperature coolant to the molecular sieve.

The molecular sieve requires 600 Btu/hr at a coolant inlet temperature of 125 F for its desorption cycle for normally efficient operation. The Block II type cold plate network does not heat the coolant temperature sufficiently high for molecular sieve use, therefore, several methods of providing this high temperature to the molecular sieve were investigated: (1) add a coolant heater to the outlet side of an essentially Block II cold plate network, (2) add a coolant heater to the outlet of a completely redesigned Block II cold plate network, (3) obtain the desired temperature by completely redesigning the cold plate network and operating the inverters at their maximum design temperature of 150 F, and (4) provide a heating system for the molecular sieve which is independent of the cold plate system except for its coolant supply, in which case the main cooling system serves as a reservoir for the molecular sieve heating system.

A schematic diagram for method 1 is shown in Figure 19. For a minimum expected cold plate system heat load condition of 350 watts, the coolant temperature is raised approximately 8 F, or from 53 F to 61 F. The heater must raise the temperature to 125 F. Assuming that 25 pounds

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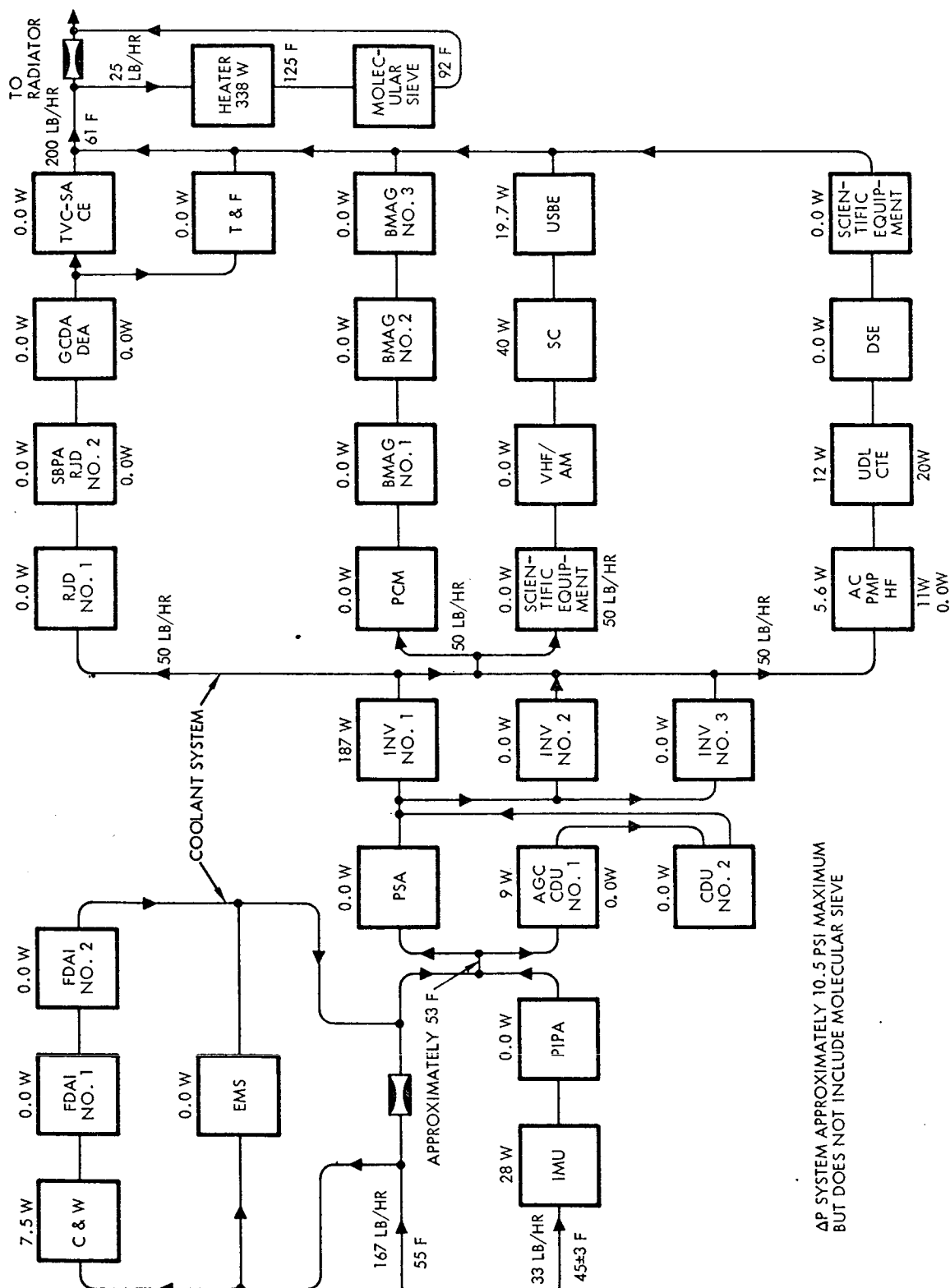
Figure 19. Cold-Plate System Schematic, Concept 1, Minimum Thermal Load Conditions

ΔP SYSTEM APPROXIMATELY 10.5 PSI MAXIMUM BUT DOES NOT INCLUDE MOLECULAR SIEVE

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per hour is needed for the molecular sieve to keep the outlet temperature high enough for normally efficient operation, the power requirement for the heater becomes 338 watts maximum. For an average cold plate system heat load condition of 431 watts, the average power requirement for the heater becomes 327 watts for a flow of 25 pounds per hour to the molecular sieve. The coolant outlet temperature from the molecular sieve for these conditions will be approximately 92 F. Other conditions of temperature and flow rate can be chosen for the molecular sieve, however, its efficiency is increased when operating at higher temperatures.

From the ECS viewpoint, method 1 has the least impact on the ECS, requiring only the addition of a heater to an essentially Block II cold plate network. The addition of two redundant heaters should be considered for mission success reliability. From a power consumption viewpoint, however, the added average drain of 327 watts on the fuel cells may present an additional cryogenics requirement. On the other hand since only 176 watts of this power is used in the molecular sieve, the remaining 151 watts can be deducted from the radiator heater requirements during low thermal loads and is not necessarily wasted. However, since the molecular sieve will present a constant power requirement even when the radiator heaters are turned off, some power will be wasted which will average less than 151 watts. The best overall solution to this problem will have to be studied during the next phase of AES.

Method 1 requires heating 25 pounds per hour of fluid from 61 to 125 F, an average consumption of 327 watts. In the molecular sieve, the fluid is cooled from 125 to 92 F, which is equivalent to 176 watts. The difference, 151 watts, is removed by the radiator, which means that the radiator heater requirement can be reduced by 151 watts in the low load condition. The total molecular sieve penalty, using the basic preliminary figures from the CO_2 Removal section is:

Molecular sieve weight	93 pounds
Net heater power penalty, 176w.	323
	416 pounds

ECS RADIATOR PERFORMANCE EVALUATION

The ECS radiators reject the excess heat produced by the equipment and astronauts in the CM and the energy absorbed by the CM from the external environment such as the sun, moon, or earth. The two Block II radiators are on opposite sides of the SM circumference. Each radiator has an area of 50 square feet and subtends an angle of about 127 degrees. The secondary coolant loop uses portions of the primary radiator system as its radiator.

The radiators use the concept of selective stagnation. This is a method of automatically varying the area of the radiator with heat rejection rate; it permits the radiator outlet temperature to fluctuate over a more narrow range than if there were no area control. The radiator system also includes a bypass circuit which operates at low heat loads to bypass fluid around the radiator, allowing lower heat rejection rates than would be possible with no bypass. A proportioning valve is used to apportion the flow between the two radiators. One radiator may experience a lower external environmental heat load than the other radiator, depending on the attitude of the spacecraft in relation to the earth, sun, and moon. The proportioning valve directs the larger portion of the coolant flow to the radiator having the lower environmental heat load.

With these three types of controls (selective stagnation area control, radiator bypass control, and proportioning valve control) the radiators will reject heat over the major portion of the range of heat loads expected for Block II missions, but will not handle the extremes of the heat rejection range. At the upper end of the range a water boiler is used in conjunction with the radiators to dissipate the required heat. At the lower end an electrical heater is used to heat the coolant as a substitute means of keeping the heat rejection rates above the minimum value the radiator is capable of handling. If the heater is not used, the radiator would freeze and become inoperable.

The Block II radiators were analyzed, from available data, to determine their performance over the range of heat rejection loads predicted for the AES reference missions and to determine problem areas which require further investigation. This study was made in three parts: (1) an investigation of the energy absorbed by the radiators from the external environment, (2) compilation of the expected heat rejection loads for the AES reference missions, and (3) determination of the conditions which define the requirements for water boiling or for the use of a heater. Problem areas of major

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significance are: (1) generally lower heat rejection loads than Block II, possibly requiring some redesign of the radiator system, (2) increased meteoroid protection requirements for extended missions, and (3) the possibility of degradation of the radiator coating in the longer missions due to nuclear bombardment.

The supplementary heater in the coolant loop to prevent the radiator from freezing presents a large launch weight penalty. Depending on the mission, up to 170 pounds of hydrogen and oxygen may be required for the fuel cells to supply the amount of make-up energy required at the predicted heat loads. This suggests the possibility of redesign of the radiator to accommodate the heat load range and provide a more versatile ECS. The redesign of the radiator would probably impose a weight penalty in itself but this penalty should be far less than 170 pounds. However, any definition of proposed radiator changes must be preceded by further testing of the present radiators to define actual radiator performance and by further investigation of applicable systems.

Additional gas storage tanks must be placed in the SM. To accommodate these modifications, the radiator inlet and outlet manifolds may be displaced slightly from their present positions. Although the change does not require major modification of the equipment, it is a significant effort. The manifolds largely determine the radiator performance characteristics with the selective stagnation concept. Therefore, not only the effect of the change in the manifold configuration, but also the thermal effect of the storage tanks must be considered to ensure that the proper radiator performance is maintained.

With the increase of mission times up to 45 days, the problem of meteoroid protection for the radiators must be considered. The thickness of the outer portions of the radiator tubes for the present Block II vehicles will not be adequate for the longer flights. If the reliability considerations are apportioned as they presently are for Block II, the outer face sheet thickness must be increased to approximately 0.305 inch. The present thickness is 0.066 inch. The 0.305 inch far exceeds the capability of the present radiator fabrication process. However, the figure was based on a first-cut minimization of the spacecraft weight to obtain the required vehicle reliability. It may be possible to compromise this figure by reapportioning the reliabilities and still maintain the overall reliability. On the other hand, it may be necessary to change the fabrication process to accommodate the larger thickness.

Another factor to be considered is the possibility of degradation of the radiator surface coating in deep space environments due to nuclear . bombardment. Although data with which to evaluate the effect of these incident particles are very scarce, preliminary information indicates that

prolonged exposure to such an environment for the length of the expected missions will cause a pronounced increase in the absorptivity of the radiator coating. Even a small increase in the absorptivity causes an important decrease in the heat rejection capability. It should be noted that this problem is not peculiar to the radiator, but will be experienced by the vehicle as a whole; the CSM and radiator coatings will be affected in the same manner, although probably not to the same degree.

The conclusions concerning the AES radiators are as follows:

1. The most energy absorption by the radiators from the external environment will be encountered in earth polar orbit, with the apex to the earth's surface, and the vehicle passing along the terminator.

2. In earth orbit the most energy is absorbed in polar orbits which pass along the terminator, while in lunar orbit maximum energy absorption occurs in orbits passing through the subsolar point.

3. The spacecraft attitude which causes the most energy absorption is generally that of the X-axis parallel to the velocity vector. However, in earth polar orbit the case of apex to the earth with the orbit along the terminator causes a high energy absorption.

4. The change in earth orbit from 100 to 200 nautical miles causes less than a 5 percent decrease in the radiator energy absorption from external sources.

5. Earth synchronous orbit missions can be considered as equivalent to cislunar because the effect of the earth emission and earth reflected solar energy is small.

6. In general the energy absorbed by the radiators in earth polar orbit is greater than in low inclination earth orbit but is less in lunar polar orbit than in lunar low inclination orbit.

7. Radiator heat rejection loads are lower than those for Block II in all phases of the mission. In effect, the radiators are too large and are susceptible to freezing.

8. In many phases of the missions the average radiator heat rejection loads are below the minimum load at which the radiators will operate. The heater in the coolant loop to prevent the liquid from freezing will remain in operation for long periods of time.

9. The coolant heater used in Block II is not adequate for AES missions and must be increased in size unless the radiators are redesigned.

10. Low inclination lunar orbit missions require the maximum use of supplementary heating and boiling.

11. The additional gas storage tanks in the SM may require relocation of the radiator inlet and outlet manifolds. These must be displaced slightly toward the vertical centerline of each radiator panel, but the relocation may be accommodated so it will not affect the performance of the radiators.

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RECOMMENDED AES-ECS SYSTEM

The recommended ECS for both the 45-day earth orbital mission and the 34-day lunar orbital mission of the AES is shown schematically in Figure 20. The system does not include redundant components other than those already present in the current Apollo Block II ECS. Redundancies shown within dashed lines are inherent in the individual components (i.e., they are built-in redundancies and do not indicate duplicate items).

The AES will utilize a two-gas system consisting of 70 percent oxygen and 30 percent nitrogen diluent. Total pressure for the AES system is 5.0 psi. This compares to the Block II 5.0 psi system consisting of oxygen only. Hardware changes involve an additional diluent supply system and an oxygen partial pressure control unit.

Block II utilizes lithium hydroxide canisters for CO_2 removal with the crew in their suits, in the cabin, or any combination thereof. Due to the excessive weight and volume involved in the storage of LiOH canisters for the longer AES missions, the AES CO_2 removal system will utilize a molecular sieve for all operations with crew in the cabin. Any time one or more crew members are suited, LiOH will be used. Except for mission abort, which is based on all crew members in their suits, mission time utilizing LiOH is considered to be a small percentage of the total mission duration. Some additional hardware such as compressors, valves, and plumbing are required to integrate the molecular sieve into the ECS.

The requirement for providing atmosphere composition control necessitated the installation of a fan for intermodule atmosphere circulation. Details of design, performance, and equipment selection were discussed in the Atmosphere Interchange section of this document.

ITEM LIST FOR FIGURE

ITEM NO.	DESCRIPTION
	ATMOSPHERIC CONDITIONING AND SUIT LOOP
1.3	Suit return check valve
1.8	Debris trap
1.10	Suit compressor
1.11	Suit compressor check valve
1.14	CO₂ absorber element
1.15	CO₂ absorber canister
1.16	Suit bypass valve
1.17	Molecular sieve shutoff valve
1.21	Charcoal filter element
1.29	Suit heat exchanger assembly
1.31	Suit flow limiter
1.32	Suit hose connector
1.33	Suit hose connector
1.34	Suit hose connector
1.36	Cyclic accumulator valve assembly
1.38	Cyclic accumulator valve control
1.39	Cyclic accumulator oxygen warning sensor
1.40	Cyclic accumulator oxygen signal conditioner
1.41	Temperature sensor
1.42	Temperature control
1.43	Suit evaporator back-pressure valve
1.44	Wetness control
1.45	Glycol temperature sensor
1.46	Glycol bypass valve
1.48	Switch
1.49	Temperature sensor
1.50	Glycol bypass control
1.51	Molecular sieve canister
1.52	Gas valve assembly
1.53	Timer and valve control
1.54	Glycol valve assembly
	COOLANT LOOPS
2.1	Space radiator check valve
2.2 2.3	Glycol pressure relief valve
2.5	Glycol check valve
2.6	Glycol evaporator
2.7	Secondary glycol evaporator
2.13	Cabin temperature control valve
2.20	Radiator control valve
2.23	Glycol temperature sensor
2.24	Glycol fill connection
2.28	Glycol shutoff valve

ITEM NO.	DESCRIPTION
2.29	Glycol reservoir
2.30	Glycol pump assembly
2.33	Glycol check valve
2.35	Secondary cabin heat exchanger glycol bypass valve
2.39	Back-pressure control valve
2.40	Back-pressure control
2.43	Wetness control
2.44	Wetness sensor
2.46	Glycol shutoff valve
2.47	Glycol temperature sensor
	ATMOSPHERIC THERMAL CONTROL
3.1	Cabin pressure relief valve
3.2	Cabin heat exchanger
3.5	Temperature selector
3.6	Cabin temperature anticipator
3.7	Temperature control
3.8	Cabin temperature sensor
3.14	Cabin fan closure
3.18	Cabin blower
3.28	Cabin pressure regulator
3.30	LEM pressure 3-way valve
3.31	LEM tunnel pressurization valve
3.32	LEM pressure gauge
	ATMOSPHERIC STORAGE AND SUPPLY
4.13	Backpack supply valve
4.16	Demand pressure regulator
4.17	Manual metering valve
4.22	Emergency inflow control valve
4.24	Pressure regulator assembly
4.25	Oxygen check valve
4.26	Oxygen shutoff valve
4.27	Relief valve
4.28	Depressurization shutoff valve
4.31	Filter
4.33	Backpack discharge coupling
4.34	Oxygen 3-way valve
4.35	Backpack charge coupling
4.62	Partial pressure regulator
4.64	Oxygen cryogenic heat exchanger
4.66	Partial pressure sensor
4.67	Partial pressure control
	WATER MANAGEMENT SUBSYSTEM
5.2	Water check valve
5.10	Potable water tank

ITEM NO.	DESCRIPTION
5.14	Water chiller
5.15	Waste water tank
5.19	Water tank pressure relief valve
5.22	Potable water supply assembly
5.23	Relief valve
5.24	Tank pressure control valve
5.29	Glycol evaporator water control valve
5.30	Suit evaporator water control valve
5.31	LEM water disconnect coupling
	POSTLANDING
6.1	Postlanding blower
6.2	Postlanding shutoff valve
6.4	Intercompartment blower
	INSTRUMENTATION
7.6	Pressure transducer
7.12	Differential pressure transducer
7.17	Gas temperature sensor (suit inlet)
8.1	Pressure transducer
8.16	Glycol accumulator quantity transducer
8.17	Evaporator steam duct pressure transducer
8.22	Steam duct pressure transducer
8.23	Glycol temperature sensor (radiator)
9.2	Oxygen flow transducer
9.8	Oxygen pressure transducer
9.11	Cabin temperature sensor
10.5	Cabin pressure transducer
11.2	Waste water tank quantity transducer
11.10	Potable water tank quantity transducer
12.1	Temperature transducer amplifier power supply
13.1	Signal amplifiers
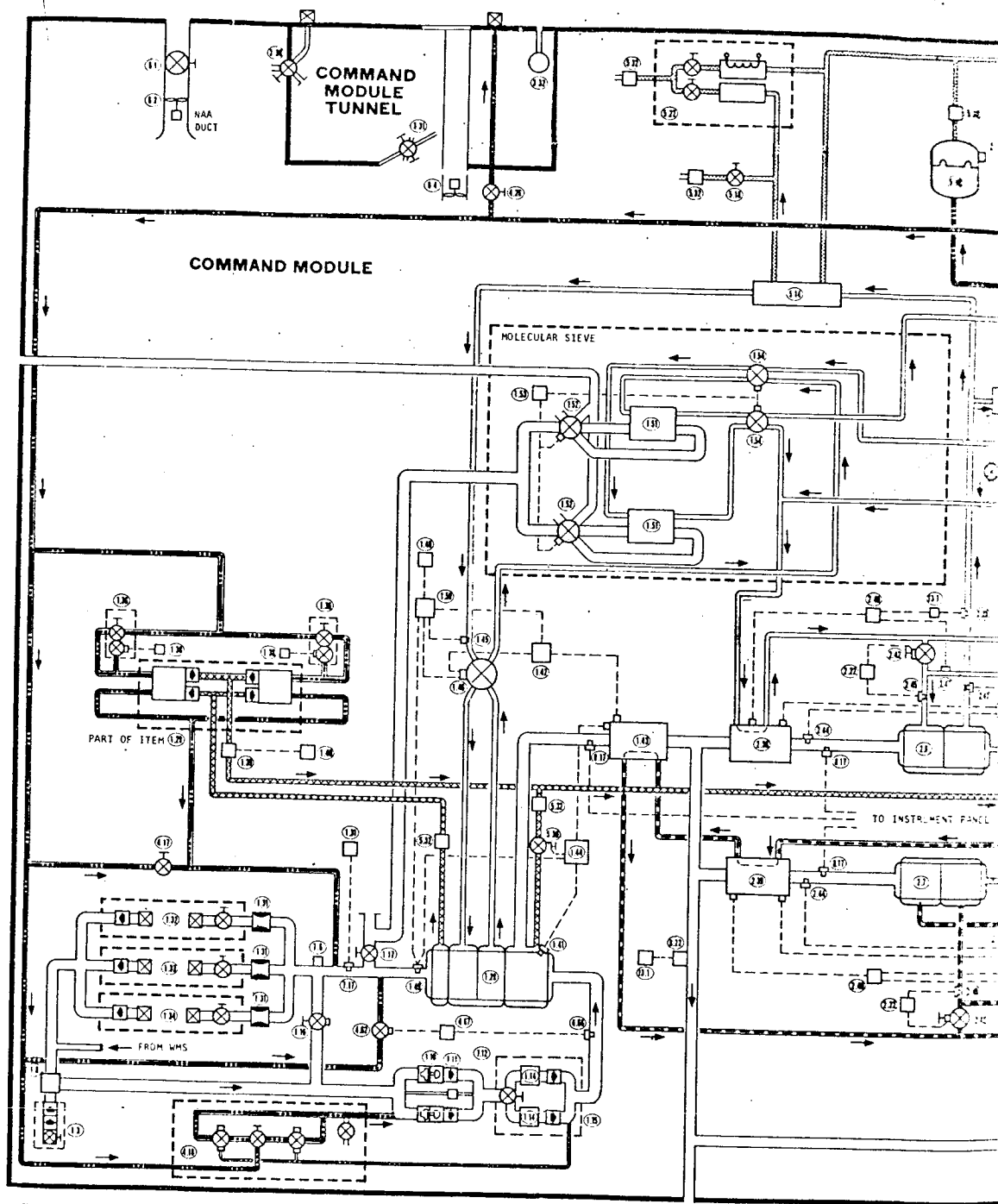
⊗ MANUAL SHUTOFF VALVE

⊗ ELECTRICALLY ACTUATED VALVE

⊗ ELECTRICALLY ACTUATED CONTROL VALVE WITH MANUAL OVERRIDE

COMMAND MODULE TUNNEL

NAA DUCT

COMMAND MODULE

MOLECULAR SIEVE

PART OF ITEM

FROM WMS

TO INSTRUMENT PANEL

MANUAL SHUTOFF VALVE

ELECTRICALLY ACTUATED VALVE

ELECTRICALLY ACTUATED CONTROL VALVE WITH MANUAL OVERRIDE

PRESSURE REGULATOR

PRESSURE REGULATOR AND RELIEF VALVE

PRESSURE RELIEF VALVE

PRESSURE RELIEF VALVE WITH MANUAL OVERRIDE

ELECTRICALLY ACTUATED MULTIPORT VALVE

ELECTRICALLY ACTUATED MULTIPORT VALVE WITH MANUAL OVERRIDE

MANUAL MULTIPORT VALVE

CHECK VALVE

SELF-SEALING QUICK-DISCONNECT

LEGEND: SECONDARY GLYCOL

POTABLE

WASTE
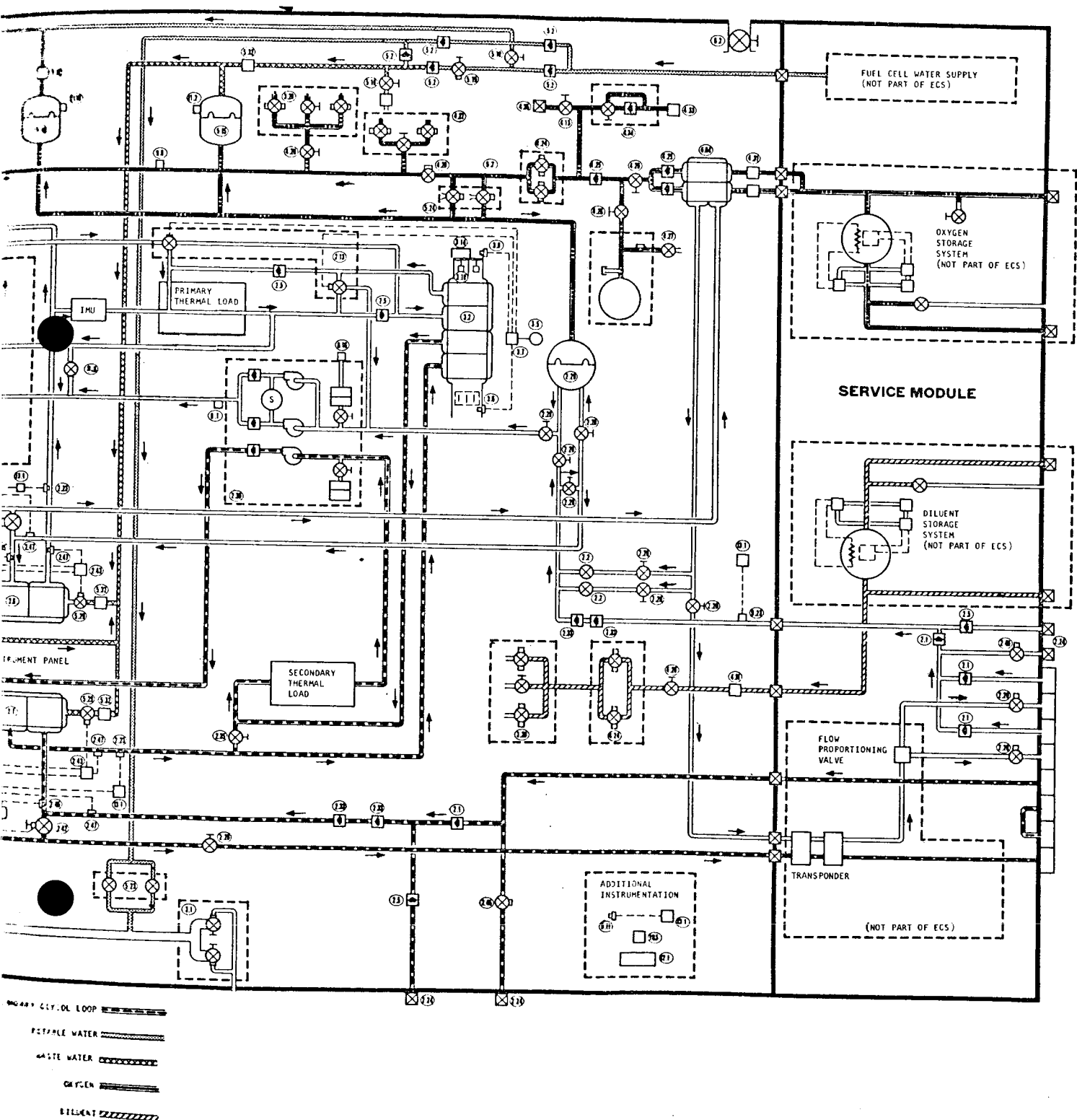
Figure 20.   Recommended ECS Schematic

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ECS CONCLUSIONS AND RECOMMENDATIONS

The preceding sections have presented a study based on the capability of a modified Block II ECS to support the extended mission requirements of the AES program. From the results of the analyses, it is concluded that with the necessary modifications, the Block II ECS concept is adequate for the AES. The modifications found necessary affect the atmosphere supply and control system, the CO_2 removal system (and as a direct result, the suit circuit), atmosphere circulation, and to some extent, the radiator operation.

Investigation showed that it is feasible and practical to modify the atmosphere supply system so that the crew can select either a pure oxygen atmosphere or a mixture of oxygen and nitrogen at a total pressure of 5.0 psia. The additional components required are a nitrogen total pressure regulator, an oxygen partial pressure sensor, controller, and a partial pressure regulator. Integration of these components with the existing oxygen supply system is quite simple.

The preliminary trade-off studies showed a distinct advantage of a two bed thermal swing molecular sieve over LiOH for CO_2 removal and control. Further investigations on the integration of the molecular sieve into the CM, which delineate the requirements for thermal control and vacuum desorption, have pointed out some major problems which require further study.

Analysis of the modified suit circuit incorporating the molecular sieve showed that both suited and shirtsleeve operations can be satisfactorily carried out. The recommendation is for minimum modification, which requires only one change in ducting—a "tee" connection for the molecular sieve downstream of the suit heat exchanger. The suit compressors can be used without modification, recognizing the power penalty due to off-design operation, with the recommendation that a two-speed motor be studied to offset the power penalty.

Humidity, CO_2 control, and oxygen concentration for both the CM and LEM compartments can be maintained at the proper operating levels by circulation of the atmosphere between the modules. Circulation will be accomplished by using a fan and duct. The fan is the same component as the existing post landing fan, and requires no development. Circulation requirements for temperature control are not necessary, since the LEM has its own temperature control system.

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It has been shown that the normal compartment leakage will take care of all the major expected contaminants except ammonia. The latter can be removed by impregnating charcoal with phosphoric acid; the charcoal could be carried in a modified debris trap. Debris trap modification to include charcoal is also necessary because of the LiOH cartridge removal in the modified suit loop. Other contaminants, such as organic compounds, can be absorbed either physically or chemically, and it is recommended that a chemical system be studied and developed, and that a catalytic burner system be dropped from consideration.

Analysis of the Block II coldplate network showed that the existing system is adequate for AES, because the AES electronic loads are less than for Apollo. However, the system cannot produce a liquid stream hot enough for the molecular sieve without addition of an electric heater at the coldplate network outlet. The addition of the heater is recommended, to avoid extensive and costly redesign of the coldplate system.

A detailed analysis of the radiator loads showed that in all mission phases of the AES, these heat loads are significantly less than those of the Block II vehicle. Thus, the Block II radiators are too large, and subject to freezing. Therefore, it is recommended that the heater in the glycol loop be increased in capacity.

In event of glycol loop failure causing an abort from lunar orbit, with minimum fuel cell water production, it was found that 100 pounds of water is necessary for suit loop cooling. The existing water tanks have a capacity for 92 pounds, of which 36 pounds is contained in the potable water tank; hence the potable water tank should be enlarged to a capacity of 44 pounds.

Finally, an analysis of the operation of the cabin temperature control system demonstrated that the design value of 75 F ±5 atmosphere temperature can be maintained for all mission modes except one. The exception is for lunar polar orbit crossing the subsolar point with the vehicle apex down, and the +Y axis along the orbit path. This particular orientation is not expected to hold for extended periods, however. Therefore, it is, concluded that the Block II temperature control system is adequate for AES.

LIFE SUPPORT SYSTEMS

The life support system, as defined in the AES study, provides the functions necessary to sustain life, with the exception of atmosphere control. These functions include storage and preparation of food, supplies, and equipment for personal hygiene; basic first aid for medical needs; waste management; and equipment for crew operations.

The Apollo Block II life support system provisions and equipment were evaluated for use in the AES program. As a result of the evaluation, several minor changes were recommended for AES. The most significant effect of the increased AES mission duration is to require increased amounts of expendables. Differences in the AES mission plans, such as crew activities connected with experiments, affect the food requirements, as well as body cleaning expendables. In general, the concepts for providing life support functions, as developed for Block II, are adequate for AES, with minor modifications.

The Block·II food allotment is 1.5 pounds per man-day, which provides an intake of 2600 to 2800 kilocalories. Because of the increased demands on the crewman's activity expected from the experiments program, it is recommended that the food allotment be increased to 1.7 pounds per man-day, which is equivalent to 2800 to 3200 kilocalories. The same type of freeze-dry food will be used for AES and Block II; obviously, more storage volume must be made available for AES. This necessary volume is not available in the CM; therefore, it must be made available in the LEM laboratory. A second recommended modification in the food subsystem involves the equipment used for hydration of the dry food. The water hydration gun was modified to include a means for measuring the quantity of water added to the food.

It is recommended that shaving equipment, fingernail clippers, and improved dental care supplies be added to the existing Block II personal hygiene supplies for AES. A larger size towellete, or cleansing pad, is also recommended, because of necessity for improved body cleaning required by the longer-duration AES missions.

The same type of medical care equipment used on Block II Apollo is recommended for AES. The quantities of supplies in the first aid kit necessary for 45 days was calculated and presented.

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It is recommended that the urine disposal lock in the waste management system be relocated so that the wick-sponge can be periodically replaced. Installation of waste discard port was investigated and is recommended for use in AES earth orbit missions.

Crew support equipment for Block II Apollo will be used for AES without modification. It is recommended that a microfilm-type flight kit be used to present mission data, experiment data, etc. and eliminate storage requirements for bulky logs, etc.

The problem of water sterilization was only briefly investigated since it was not initially within the scope of the planned work. Several schemes for sterilization were presented. No recommendation was made because the need for water sterilization should first be definitely established.

In summary, the life support systems of Block II Apollo are adequate for AES, with additional expendables for the 45-day missions and modifications as outlined herein.

THERMAL ANALYSIS

THERMAL ANALYSIS

The objective of these analyses was to evaluate the thermal constraints imposed upon the spacecraft by AES missions. Another major goal of the study was to establish present and future thermal control problem areas and, where possible, to recommend solutions.

The thermal studies performed were concentrated in the following general areas: (1) definition of the external environment, including effects of the LEM laboratory and thermal coating degradation; (2) overall command module thermal analysis; (3) overall service module thermal analysis; (4) detailed analysis of the CM reaction control system; (5) detailed analysis of the SM reaction control system; (6) preliminary analysis of the service propulsion system; and (7) detailed analysis of the parachute compartment. In conjunction with these studies, effort was directed toward summarizing the attitude constraints imposed upon the spacecraft by thermal requirements.

The general objectives of each of the above areas of study were to define problem areas, determine future analytical requirements, and evaluate thermal constraints. In addition, each area had specific objectives which were unique to that area, e. g., the major objective of the CM thermal study was to establish cabin heat rejection requirements. Similarly, the objective of the SM RCS study was to determine the adequacy of the Block II RCS heater sizes for AES missions.

For more detailed descriptions of the thermal studies, see reports SID 65-1524-1 and SID 65-1524-2.

EXTERNAL ENVIRONMENT

The definition of the external environment was based on the four AES reference missions. Likely vehicular attitudes with respect to the earth, moon, and sun were considered and compared for thermal extremes. A parametric variation of missions, with inertial and local spacecraft attitude holds, was established. Incident environmental heat loads (comprised of direct and reflected sunlight and planetary emission) were determined for the mission-attitude matrix. Tables 33 and 34 list the mission and attitude characteristics for earth and lunar missions, respectively.

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Table 33. Earth Orbits and Vehicle Orientation

Earth Orbit	Altitude (Nautical Miles)	Inclination	Day of Year	Right Ascension of Ascending Node	Eccentricity	Vehicle Orientation
1. Polar (subsolar)	200	90	80 (Vernal Equinox)	0	0	Earth oriented +X along orbit path +Y down local vertical
2. Polar (subsolar)	200	90	80	0	0	Earth oriented +X down local vertical +Y along orbit path
3. Polar (subsolar)	200	90	80	0	0	Sun oriented +X perpendicular to sun +Y toward sun
4. Polar (subsolar)	200	90	80	0	0	Sun oriented +X toward sun +Y in equatorial plane
5. Polar (terminator)	200	90	80	90	0	Earth oriented +X along orbit path +Y down local vertical
6. Polar (terminator)	200	90	80	90	0	Sun oriented +X toward sun +Y in equatorial plane
7. Synchronous equatorial (subsolar)	19321	0	80	0	0	Earth oriented +X along orbit path +Y down local vertical
8. Synchronous equatorial (subsolar)	19321	0	80	0	0	Sun oriented +X perpendicular to sun +Y toward sun
9. Synchronous equatorial	19321	0	222	0	0	Earth oriented +X along orbit path +Y down local vertical
10. Synchronous equatorial	19321	0	222	0	0	Sun oriented +X perpendicular to sun +Y toward sun

Table 34. Lunar Orbits and Vehicle Orientation

Lunar Orbit	Altitude (Nautical Miles)	Inclination	Declination[1]	Right Ascension[2] of Ascending Node	Eccentricity	Vehicle Orientation
11. Polar (subsolar)	80	90	0	0	0	Lunar oriented +X along orbit path +Y down local vertical
12. Polar (subsolar)	80	90	0	0	0	Lunar oriented +X down local vertical +Y along orbit path
13. Polar (subsolar)	80	90	0	0	0	Sun oriented +X perpendicular to sun +Y toward sun
14. Polar (subsolar)	80	90	0	0	0	Sun oriented +X toward sun +Y in equatorial plane
15. Polar (terminator)	80	90	0	90	0	Lunar oriented +X along orbit path +Y down local vertical
16. Polar (terminator)	80	90	0	90	0	Sun oriented +X toward sun +Y in equatorial plane
17. Polar	80	90	0	75	0	Lunar oriented +X along orbit path +Y down local vertical
18. Polar	80	90	0	60	0	Lunar oriented +X along orbit path +Y down local vertical

1. Angle between sun-moon line and equatorial plane
2. Measured from subsolar point

Determination of the environmental heating rates required approximating the spacecraft contoured shape as a multiplanar shape. The CM truncated cone was approximated by an eight equal-sided truncated pyramid. The SM cylinder was approximated by an eight-sided pipe.

Using a greater number of sides to the approximating figures would have resulted in a larger expenditure of computer time not only for environmental heat loads but for thermal network requirements. The error introduced by these approximations was estimated and found to be acceptable. An effective compromise between computational efforts and required accuracy was established in this manner.

OCCLUSION EFFECTS

Determination of environmental heat loads was performed with the aid of an existing computer program. However, the program theory does not provide for the possibility of a shadow being cast upon a surface by an interposing part of the spacecraft. The occlusion of the CSM by the docked LEM, and vice versa, greatly affects the environmental heat loads. Thus it was necessary to modify the incident environmental heat loads obtained from the computer program in order to include shadowing effects.

The occlusion of direct solar radiation was evaluated by revising an existing computer program. As a result, it was possible to machine calculate the illuminated area fraction of the CM for all sun-spacecraft positions of interest. Projections of the LEM-CM were produced on CRT plots which were evaluated for nonoccluded area. A typical CRT projection is shown on Figure 21. The silhouette of the crew compartment portion of an eighth CM segment is combined with the opposing LEM ascent stage. The silhouettes are produced for the view direction skewed 30 degrees from the CSM X-axis. The nonoverlapped area corresponds to the fractional CM area illuminated by the sunlight when coming from the view direction.

The occlusion of planetary emission and reflected sunlight was evaluated in another manner. Unlike collimated direct sunlight, these sources radiate diffusely, i.e., uniformly, in all directions. The interception of diffuse radiation is represented by a geometric parameter, the configuration factor. The configuration factor with LEM shadowing for all attitudes was computed with the aid of the same existing program, but employing specialized techniques.

The effect of LEM shadowing in modifying the environmental heat load is indicated in Figure 22. All LEM shadowing effects during mission case 1 of Table 33 are shown comparatively. The incident radiant heat flux during this typical orbit is shown with and without shadowing by the interrupted and continuous curves, respectively. The LEM shadowing reduces the orbital heat load to this eighth-area segment by about 35 percent.

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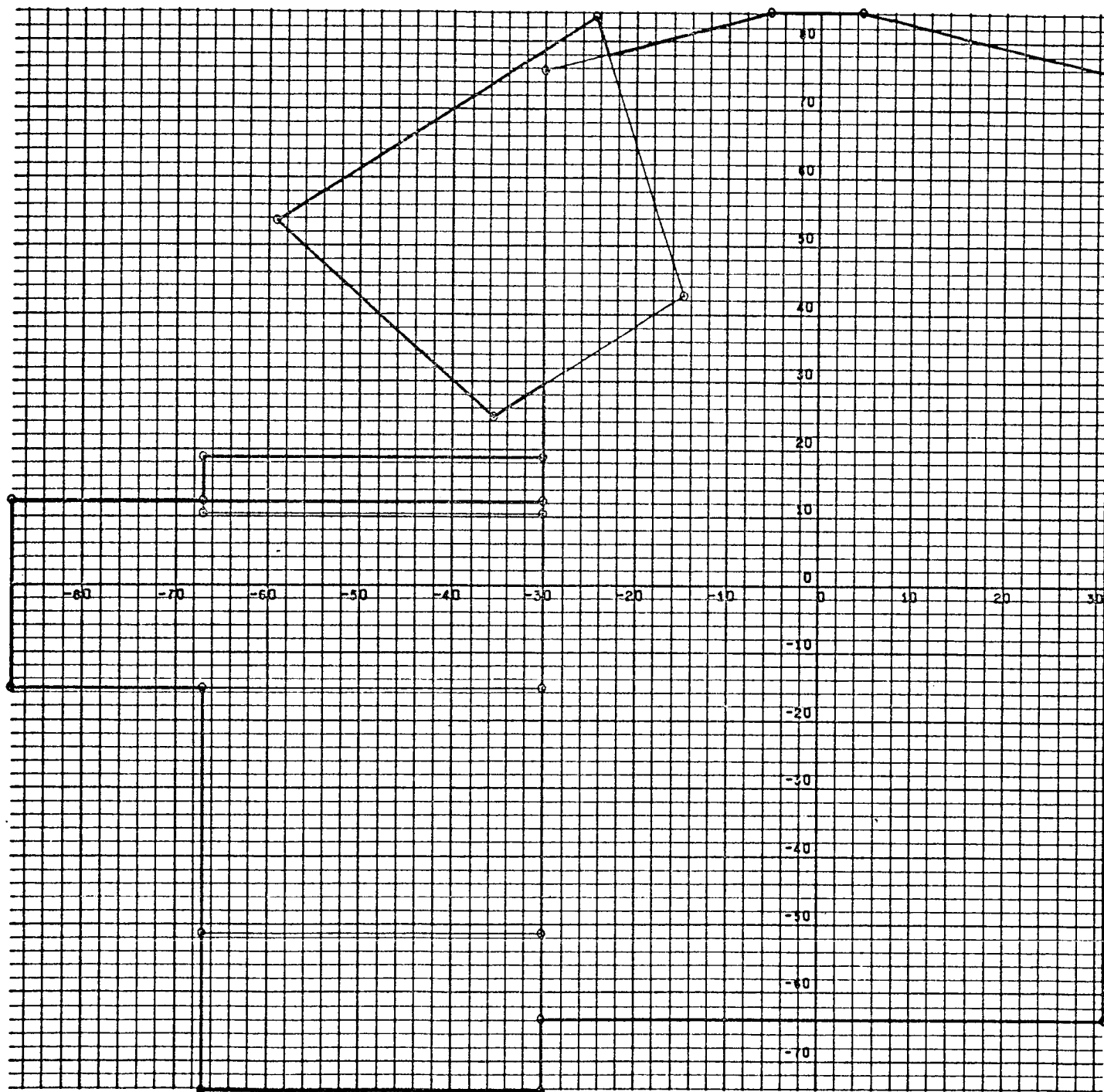
Figure 21.　View of Forward Section and Midsection, LEM Ascent Stage, Including View of Crew Compartment Portion of CM Eighth Segment No. 7
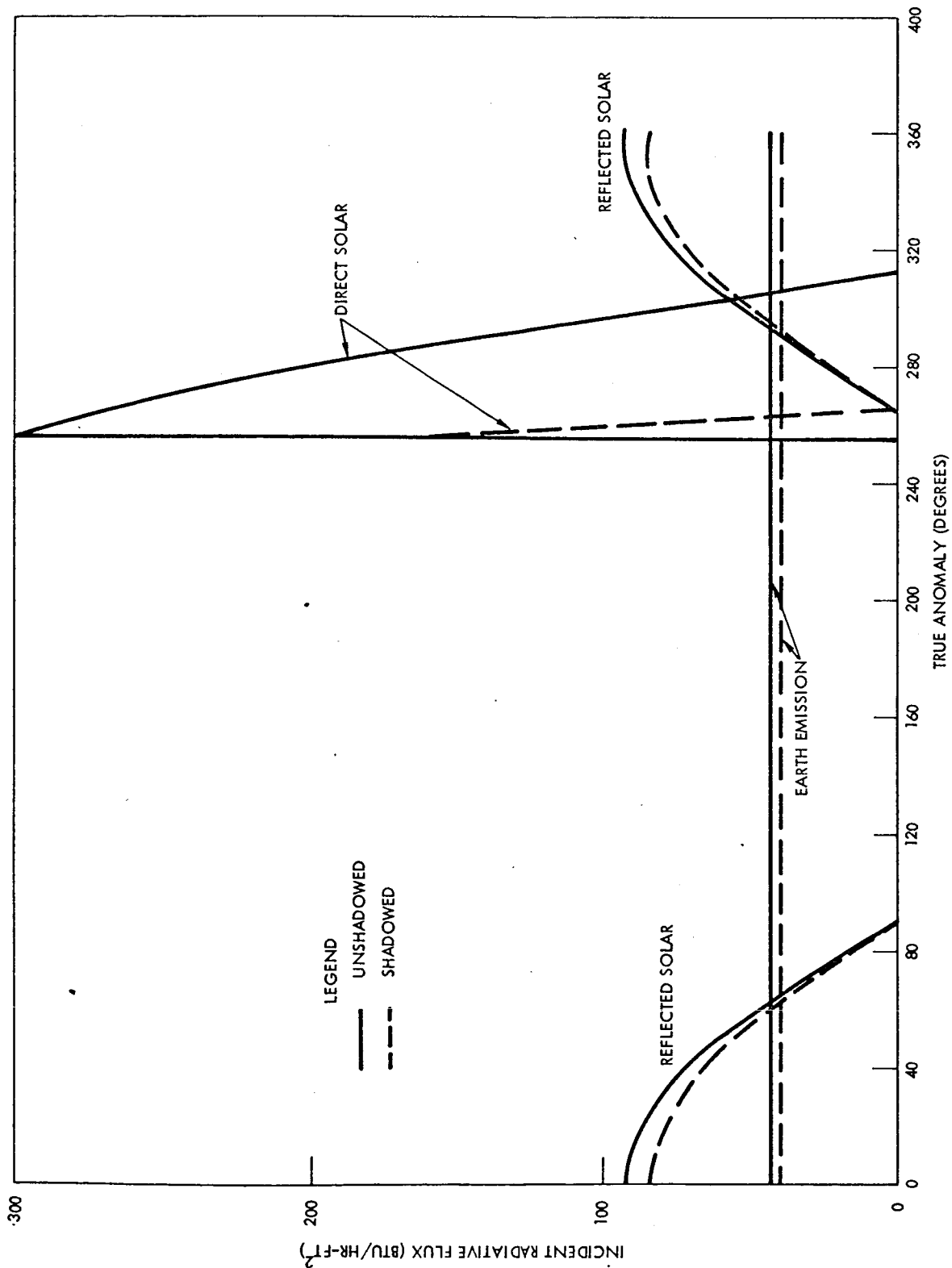
Figure 22. Incident Radiation on CM Crew Compartment, Segment No. 7, During Mission 1

INTERREFLECTION EFFECTS

The associated problem of interreflection of thermal energy by the highly reflective LEM and CM coatings was examined. The radiation incident to either structural surface is partially reflected onto the other and back again, enhancing the initially absorbed incident radiation. The enhancement of heat loads by interreflection was evaluated and found to be greatest when the LEM surface opposing the CM was in an illuminated position. The parachute compartment area of the CM was most affected. The interreflection effects were incorporated into the environmental heat loads to the thermal network of the parachute compartment. Interreflection effects were smaller elsewhere and were deleted to simplify calculations.

THERMAL COATING DEGRADATION

A review of the latest available information regarding the effects of the space environment upon thermal coating stability was made. This information consisted principally of flight data from OSO-2 and Mariner-Mars. A number of laboratory tests have indicated that a zinc oxide-potassium silicate white paint will degrade very little under prolonged ultraviolet exposure. Therefore, this type paint has been tentatively chosen for both the ECS and EPS Apollo radiators. Flight data from OSO-2 seems to verify the laboratory test data inasmuch as no detectable change in optical properties has thus far been recorded. Unfortunately, flight data from Mariner-Mars is completely contradictory to the data from OSO-2 and indicates definite degradation of the zinc-oxide paint. Admittedly, the Mariner-Mars emissivity tests were not highly sophisticated, and it is not certain whether the degradation occurred because of ultraviolet radiation, nuclear bombardment, or coating application techniques. However, it would seem imperative that there is a need for further investigation.

Nuclear bombardment, especially low-energy protons, presents a potentially serious degradation problem to not only the radiator coatings but all thermal coatings on Apollo. It was concluded that considerable investigation is needed in this area, both in defining the environment and obtaining laboratory and flight test data.

COMMAND MODULE THERMAL ANALYSIS

The thermal behavior of the command module was evaluated parametrically over a variety of natural (ambient) and induced (internal) environmental conditions. The primary results of these analyses include the estimated heat loads required for design of the environmental control system (ECS) in order to maintain appropriate cabin conditions for crew comfort. The secondary results were CM structure and component temperatures for the matrix of environmental conditions. The resultant temperature levels are effectively mean values over sizable areas. For structural and functional evaluations, the temperature levels are considered adequate; however, temperature gradients obtained may be inadequate for detailed thermo-structural analysis.

The primary results of the computations were heat production and rejection rates for acceptance by the ECS in order to maintain the specified crew cabin temperature conditions. In many cases, minimum extremes of cabin pressure-vessel wall temperature (Table 35) indicate a severe potential problem—condensation. The dew-point range of temperature is higher in some instances than that attained on the walls, making condensation likely if the atmospheric environment is suitable. Therefore, condensation continues to be a major problem for both Apollo and AES. The basic CM heat balance is shown in Figure 23, which presents the five heating rates pertaining to the gain or loss of energy by the system.

A total of 14 orbital situations was used to compare the various temperature profiles. The maximum-minimum comparison of the cabin wall temperatures and the stainless steel bond-line temperatures for each case are presented in Table 35. Typically representative results of the study are presented in graphic form in Figures 24, 25, and 26. These CRT plots represent heat shield temperature, heating rate across the heat shield, and heat sum (integrated heat) across the heat shield for an 80-nmi lunar orbit passing over the subsolar point.

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Table 35.　Command Module Maximum-Minimum Temperatures

Case No.	Cabin Wall		Stainless Steel Bondline	
	Max. °F	Min. °F	Max. °F	Min. °F
1	82	60	29	-29
2	81	60	32	-18
3	90	62	75	-32
4	70	47	11	-52
5	97	48	127	-92
6	70	47	8	-52
10A	73	59	30	-14
10B	73	58	28	-12
11	89	61	72	-31
12	87	63	52	-12
13	95	68	70	-18
14	69	42	8	-66
17	94	42	117	-84
18	90	48	93	-83
Cold* Case	57	8	-62	-143

* Cold Case – The command module external surface is
viewing deep space with no celestial bodies in view.
Internal heat generation is: (1) electrical heat
loads––922 Btu/hr and a 90 F constant temperature
equipment bay, and (2) cabin air is held constant
at 75 F. For this case, the command module network
was run with zero capacitance for all nodes giving
a steadystate equilibrium heat and temperature balance.

$Q_{ENVIRONMENT}$

① $Q_{WINDOWS}$ ③

Q_{AIR} ②

④ $Q_{ELECTRICAL}$

$Q_{NODE\ 149}$ ⑤
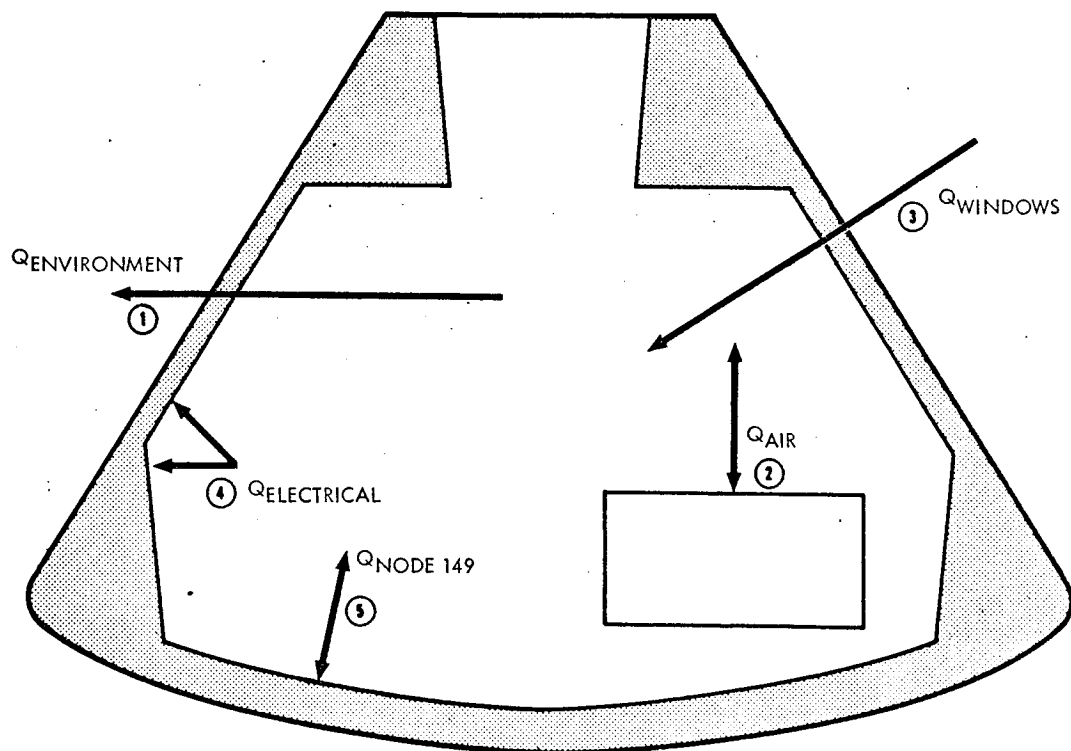
1. HEAT EXCHANGED WITH ENVIRONMENT.

2. HEATING OR COOLING REQUIRED TO MAINTAIN CABIN AIR AT SPECIFIED TEMPERATURE.

3. HEAT GAINED BY CABIN FROM WINDOWS.

4. ELECTRICAL HEAT LOAD FROM NON-COLD-PLATED EQUIPMENT INTO STRUCTURE.

5. ELECTRICAL HEAT LOAD FROM COLD-PLATED EQUIPMENT NOT REMOVED DIRECTLY BY
    EGW. HEAT LOAD SIMULATED BY A CONSTANT TEMPERATURE NODE LOCATED AT THE
    LOWER EQUIPMENT BAY RIGHT HAND.

Figure 23 .   Command Module Heat Balance

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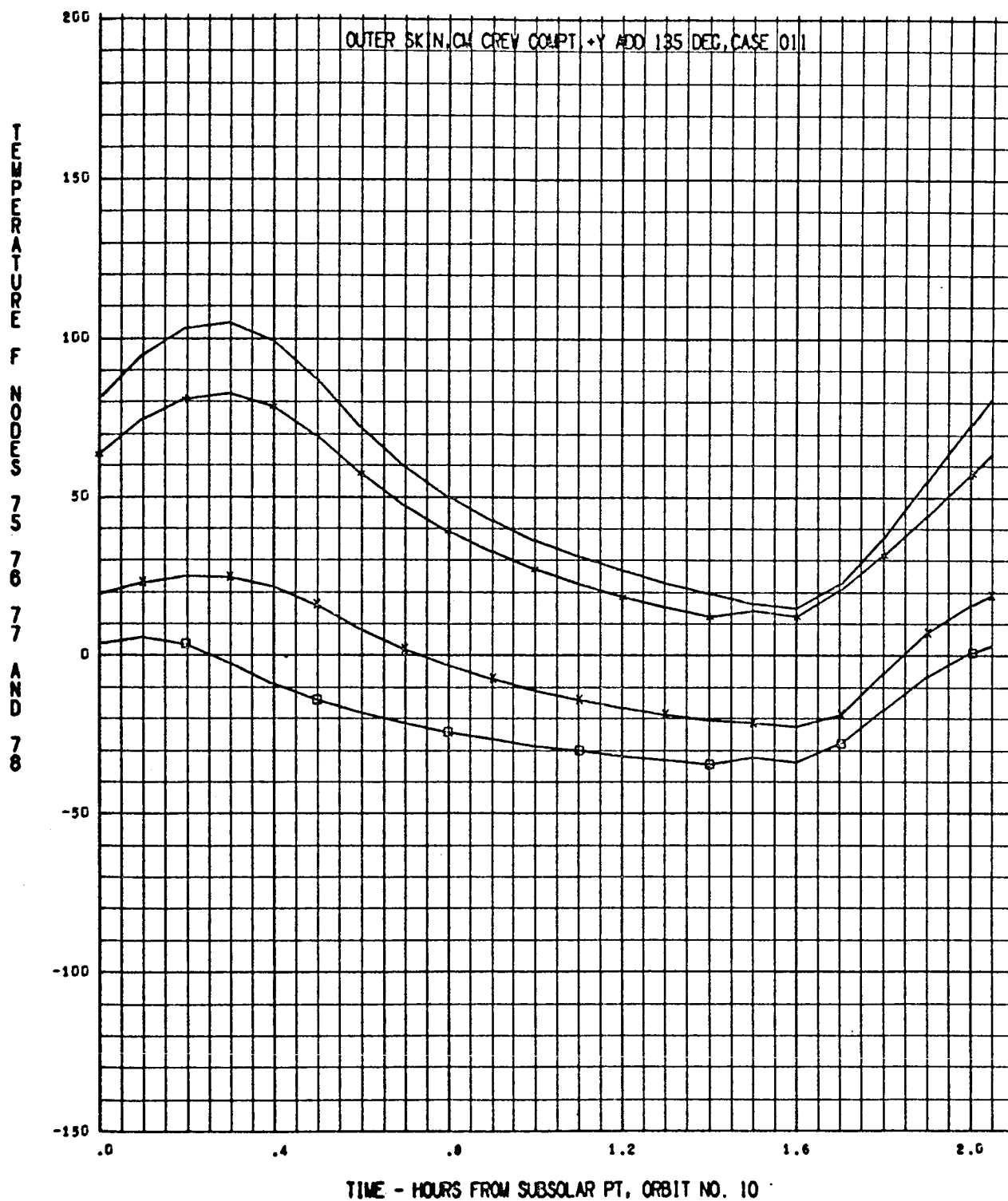
Figure 24. Heat Shield Cabin, Case 11

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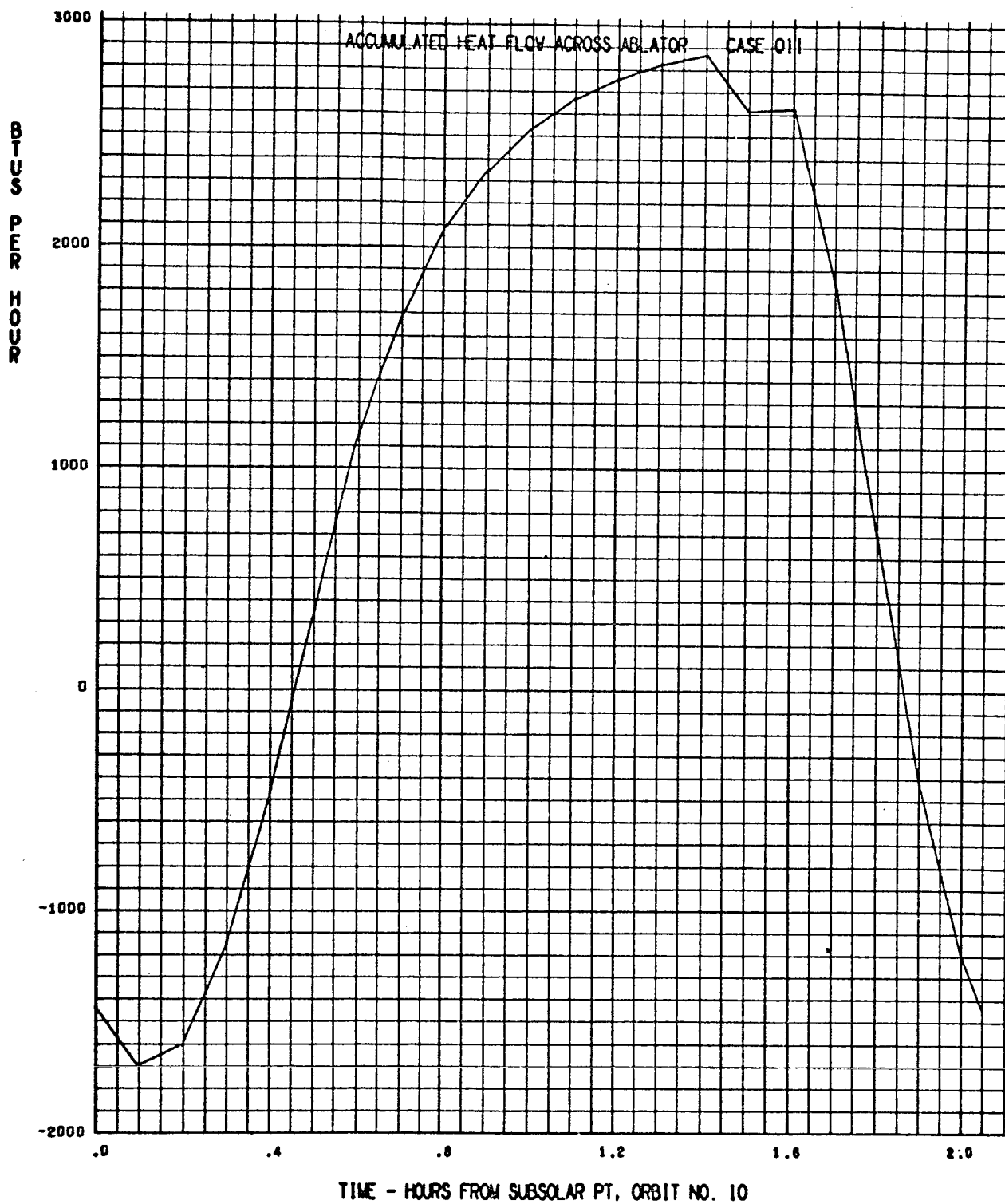
Figure 25.   Heat Rate Across Heat Shield, Case 11
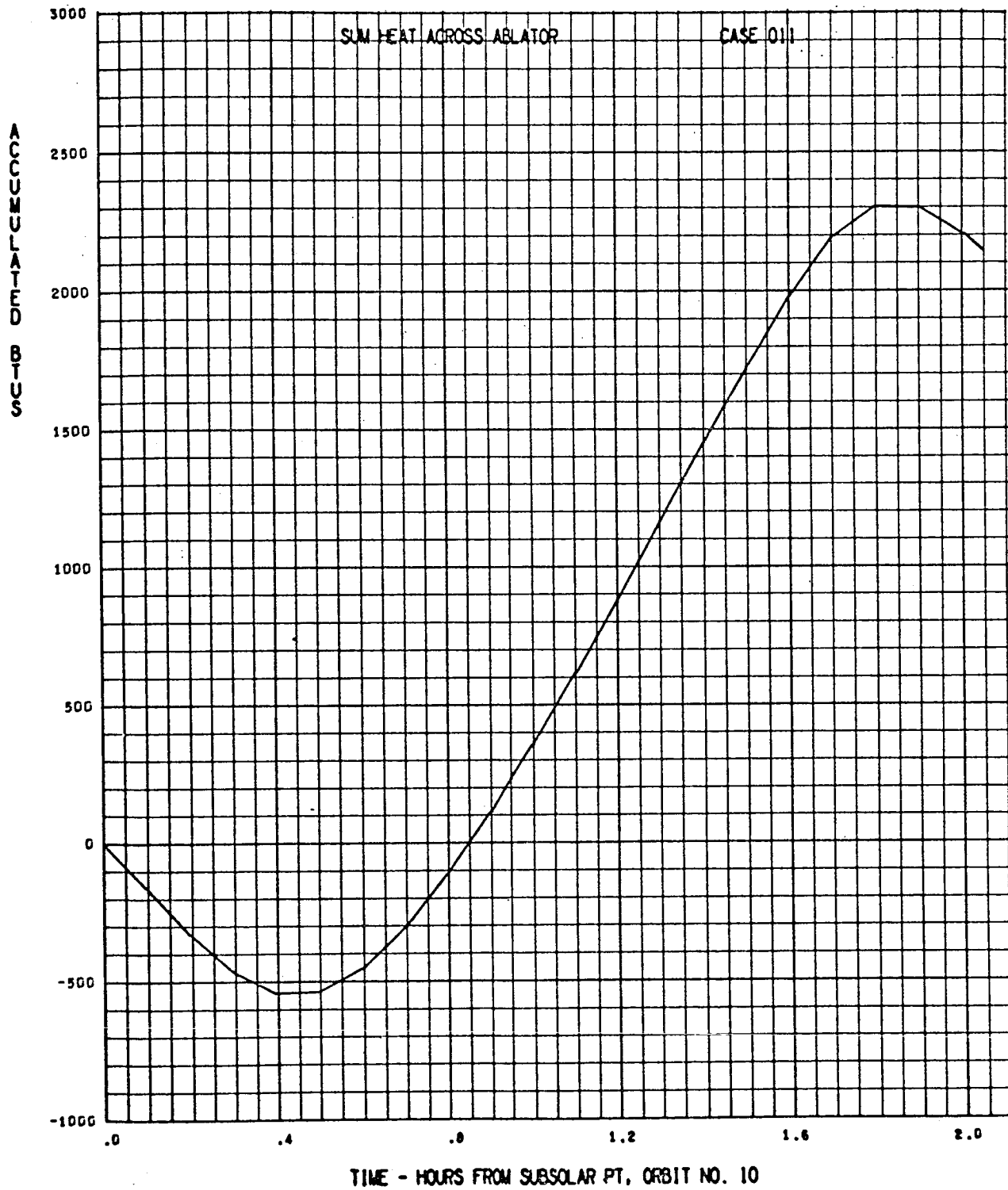
Figure 26. Heat Sum Across Heat Shield, Case 11

SERVICE MODULE THERMAL ANALYSIS

A thermal network analog of the service module was analytically constructed to calculate structural temperature distributions and histories for 14 different orbital conditions, including passive thermal control. Temperature histories of components with temperature control requirements were also calculated in order to determine the magnitude of the problems associated with meeting these requirements. AES Pratt & Whitney fuel cells and applicable Block II SM structure were represented in the thermal network analog.

To permit calculating representative temperatures in the time available, several assumptions were made in the representation within the thermal network of various components contained within the SM. A reduced mass was used in the SPS and RCS propellant tanks to permit representing each tank by one node. The fuel cells were represented as constant heat sources. The heat loss rates were taken from test data. The interiors of the cryogenic tanks were included as constant temperature nodes with a conductance to the outer surface of the tank calculated from allowable heat gains for specified surface temperatures. The effect of the SM interior insulation was represented by an equivalent emissivity assigned to all portions of the structure covered by the insulation.

Results of analyses using this network analog show that the SM exterior shell will experience temperature extremes from -160 F to +250 F, and that temperature control problems exist. Figures 27 and 28 present outer shell circumferential temperature distributions for the center nodes of the SM for earth and lunar orbits, respectively. These figures demonstrate the type of temperature fluctuations that may occur in the outer shell and are fairly representative of the structural thermal environment to which interior components will be exposed. The RCS propellant and helium tanks exceed both the upper and lower allowable temperature limits. In addition, the SPS propellant transfer lines and disconnect panels have localized temperature control problems at structural support areas. The results also show that extreme temperature distributions may occur in localized areas and may create mounting problems for components. The temperatures that result on the cryogenic tank surfaces are within allowable values; however, these results are not indicative of the temperature distributions over the entire surface. Because of the thin outer shell of these tanks, buckling due to thermal stresses may result from adverse temperature distributions.
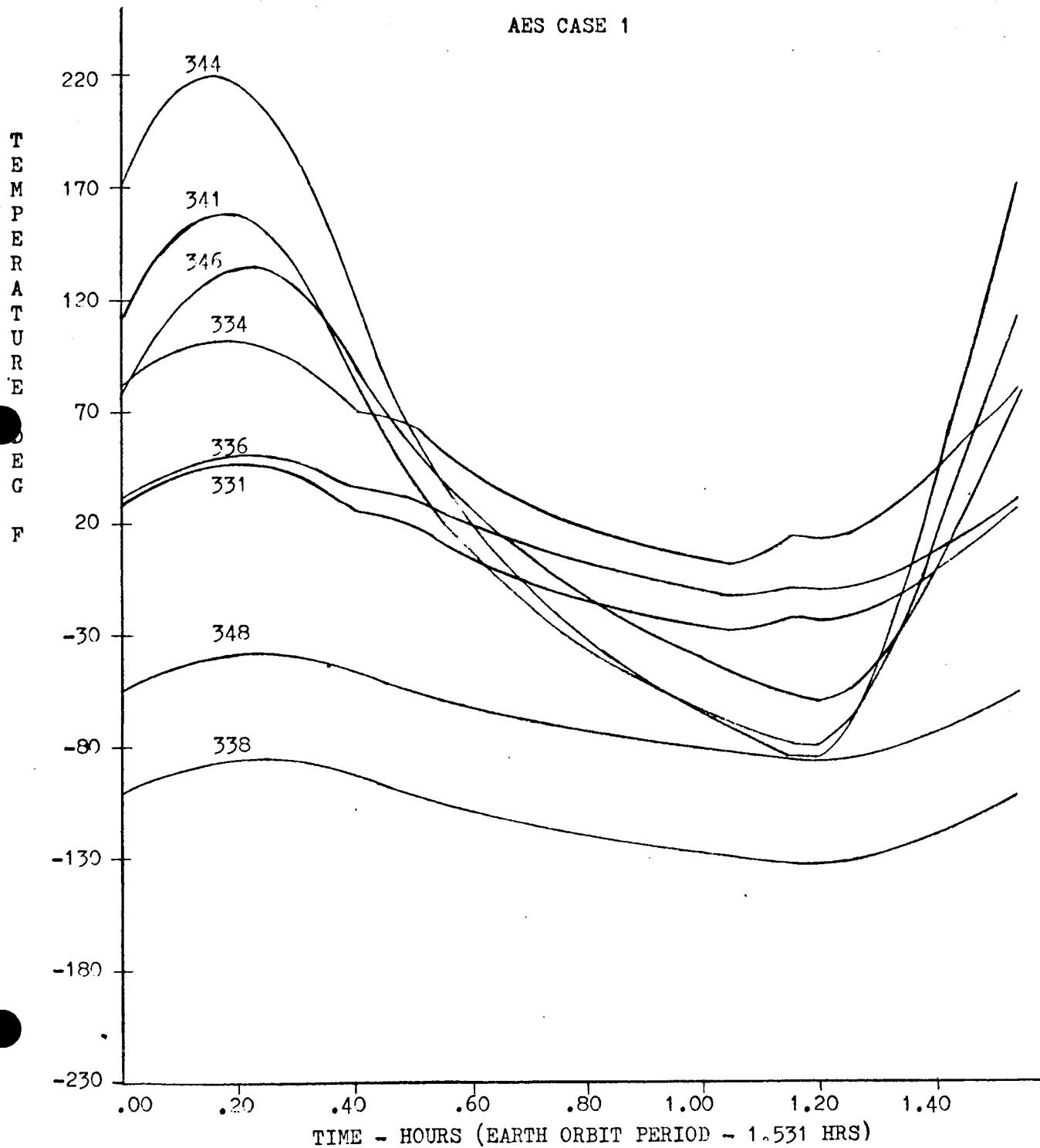
Figure 27.   Typical Outer Shell Circumferential Temperature
Distribution
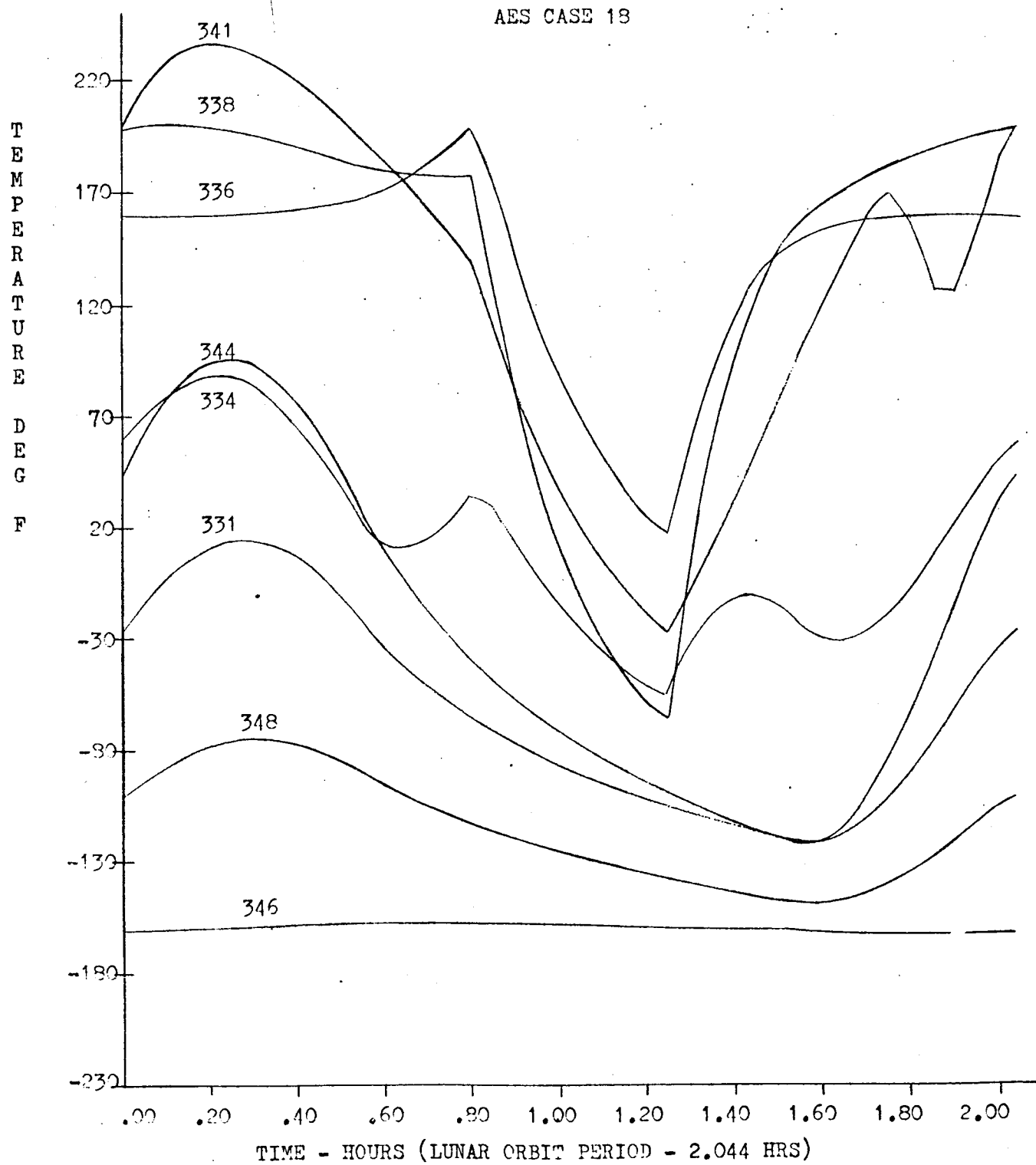
Figure 28. Outer Shell Circumferential Temperature Distribution, Case 18

The results further show that passive thermal control, as a means of providing moderate temperatures throughout the SM and maintaining temperature-sensitive components within the allowable temperature limits, is valid, except for the SPS propellant transfer lines. Figure 29 presents outer shell circumferential temperature distributions for the center nodes of the SM for one rph and demonstrates that an average temperature of 70 F results, thus providing a moderate structural thermal environment for the interior components.
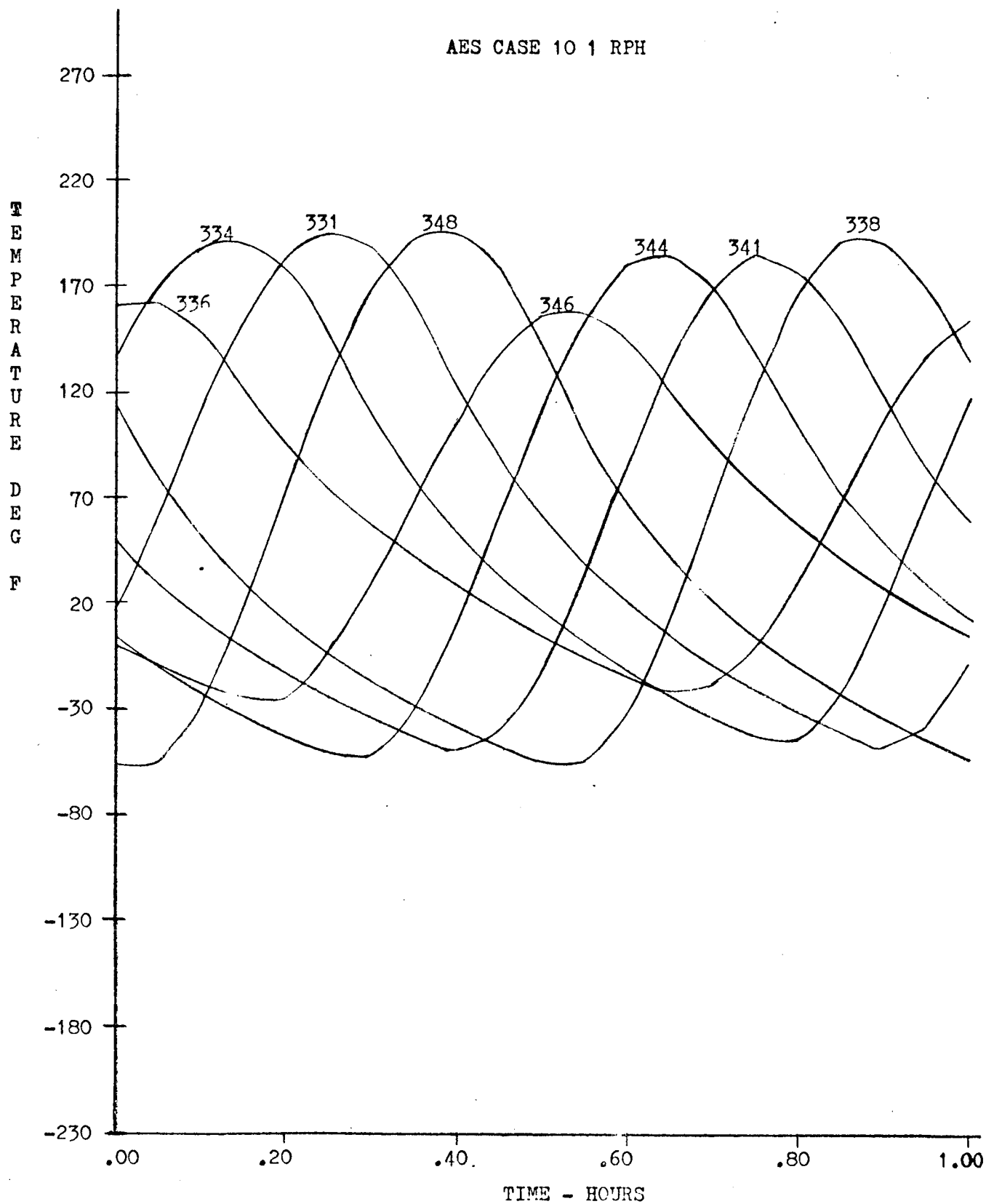
Figure 29. Outer Shell Circumferential Temperature Distribution, Case 10

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CM RCS THERMAL ANALYSIS

A thermal analysis was performed on the command module reaction control system (CM RCS) yaw engine to determine its thermal behavior for the AES missions. For all CM RCS engines, the temperature limits during all phases of any mission are -150 F to 225 F on the propellant valves (non-operating), and -200 F to 250 F on the nozzle extension (nonoperating), except prior to reentry where the temperature of the valves must be between 40 F and 100 F. Analysis of the CM RCS engine during boost and reentry was not considered in this study.

Temperature results of the yaw engine were obtained for the following environmental cases: (1) extreme cold attitude, (2) extreme hot attitude, (3) cisearth or cislunar, with PTC of 1 rph and 2.5 rph, (4) five different earth orbits, and (5) four different lunar orbits.

Figure 30 depicts a typical CM RCS yaw engine temperature history for an earth orbit. For all the AES earth orbit missions considered and for hot soak and cold soak, the propellant valves remained within their nonoperating temperature limits. However, were reentry to follow a cisearth cold soak or certain earth orbital missions, some time would be required in order to preheat the propellant valves to their lower operating temperature limit of 40 F.

A typical lunar orbit temperature history of the CM RCS engine is shown in Figure 31. The nonoperating temperature limits for the AES lunar orbital missions will be satisfied.

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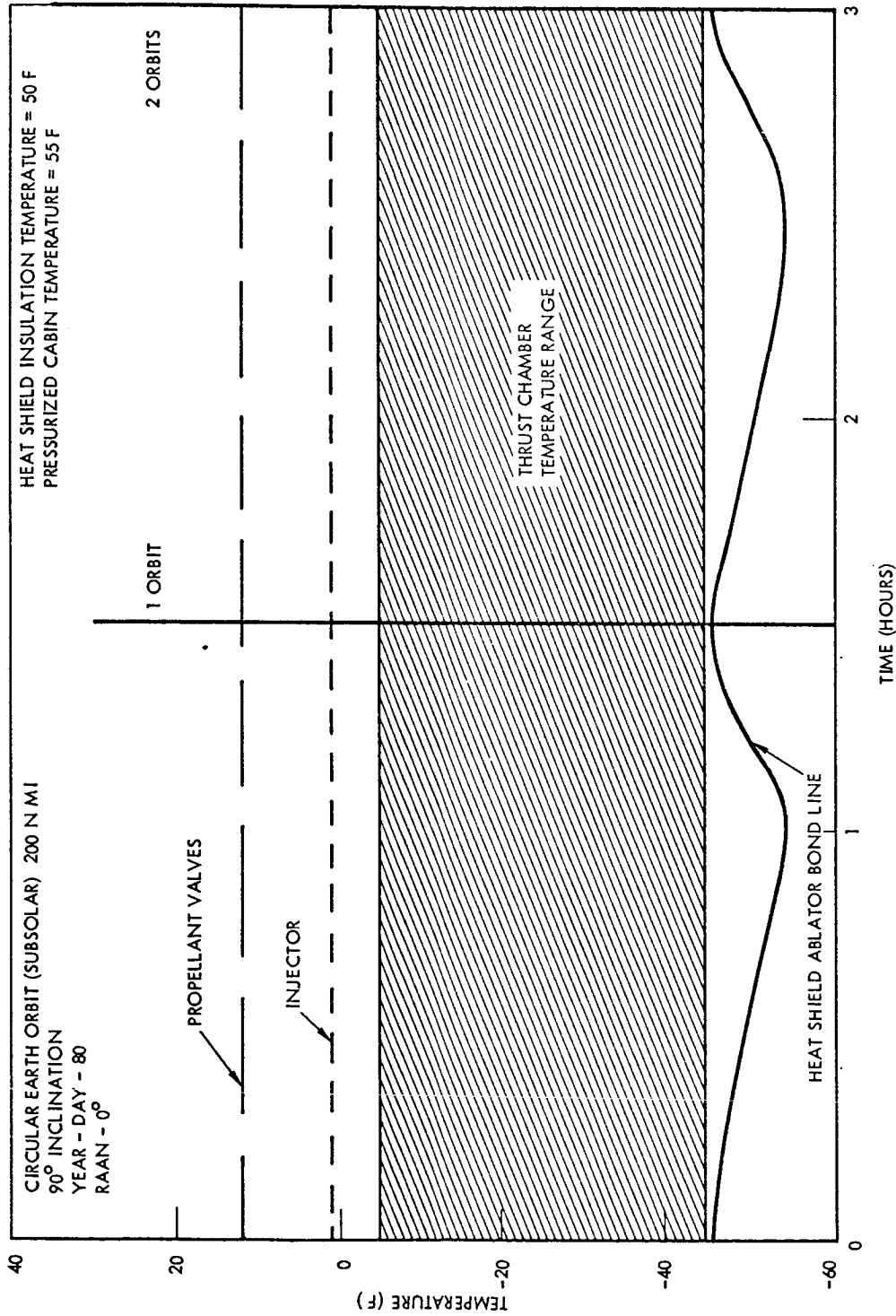
Figure 30. Temperature History of the CM RCS Yaw Engine, Case 1.

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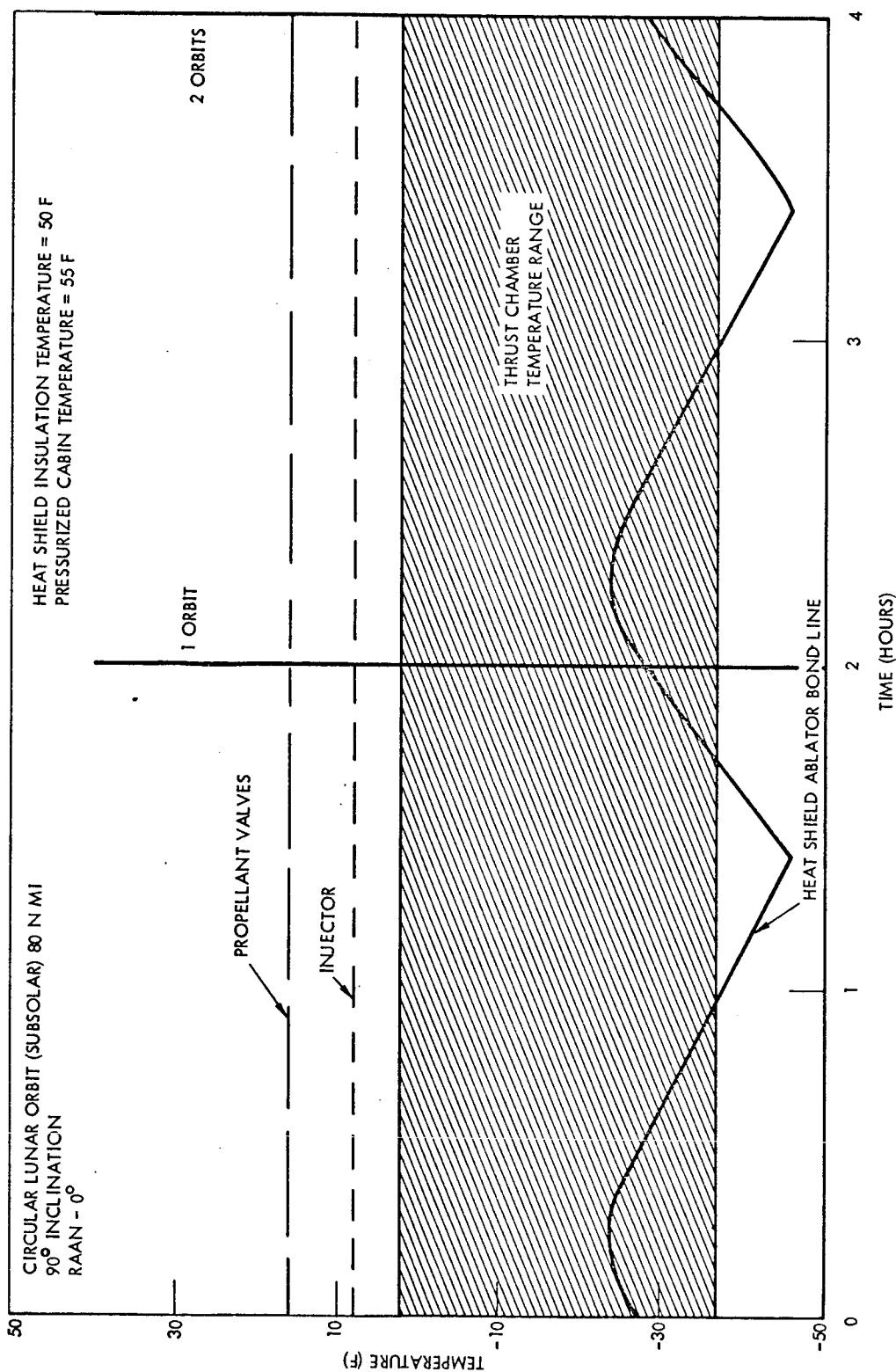
Figure 31. Temperature History of the CM RCS Yaw Engine, Case 11.

SM RCS THERMAL ANALYSIS

A thermal analysis was performed on a SM RCS quad housing and engines for the orbital and transit phases of the AES missions. Components of the SM RCS associated with propellant flow have critical temperature limits which can not be exceeded without serious hazard. Minimum allowable temperatures of the propellant valves and injectors are 35 F and 20 F, respectively; maximum allowable temperatures of the propellant valves and injectors are 175 F and 350 F, respectively. The objectives of this analysis were to determine thermal problems which might be experienced by the system, vehicle constraints, and the required heater power per quad versus cislunar attitude hold time.

Ten distinct environmental cases were investigated in this analysis: four earth orbits, five lunar orbits, and one cislunar condition. The vehicle orientation assumed for the cislunar analysis was that with the X axis held perpendicular to the solar vector. For the synchronous earth orbits, the vehicle is in the earth's shadow approximately one out of 24 hours. Hence, the conditions of the cislunar analysis are essentially the same as those that exist for the synchronous earth orbits where the X axis is perpendicular to the solar vector.

Since the major mode of heat transfer about the RCS system is due to thermal radiation, a proper accounting of all interreflections between the various surfaces was necessary in the computations. Accordingly, the assumption was made that the surfaces were gray, isothermal, and opaque, and the radiosity analog method was applied in determining the radiant interchange.

Proper definition of the environmental heating, or incident radiation, was accomplished for the cislunar analysis. However, incident radiation for the planetary analyses did not reflect the shadowing effects due to the SM, LEM, etc.

The analyses showed that, in order to satisfy the thermal constraints of the system during cislunar conditions, either PTC must be employed or a heat source for each quad must be furnished. It was determined that the minimum PTC roll rate is approximately 0.6 rph (Figure 32). The required heater power per quad for a three-hour cislunar hold was determined to be 154 Btu's per hour and 143 Btu's per hour for one rph and two rph roll rates, respectively (see Figures 33, 34, and 35). The results of the analysis also
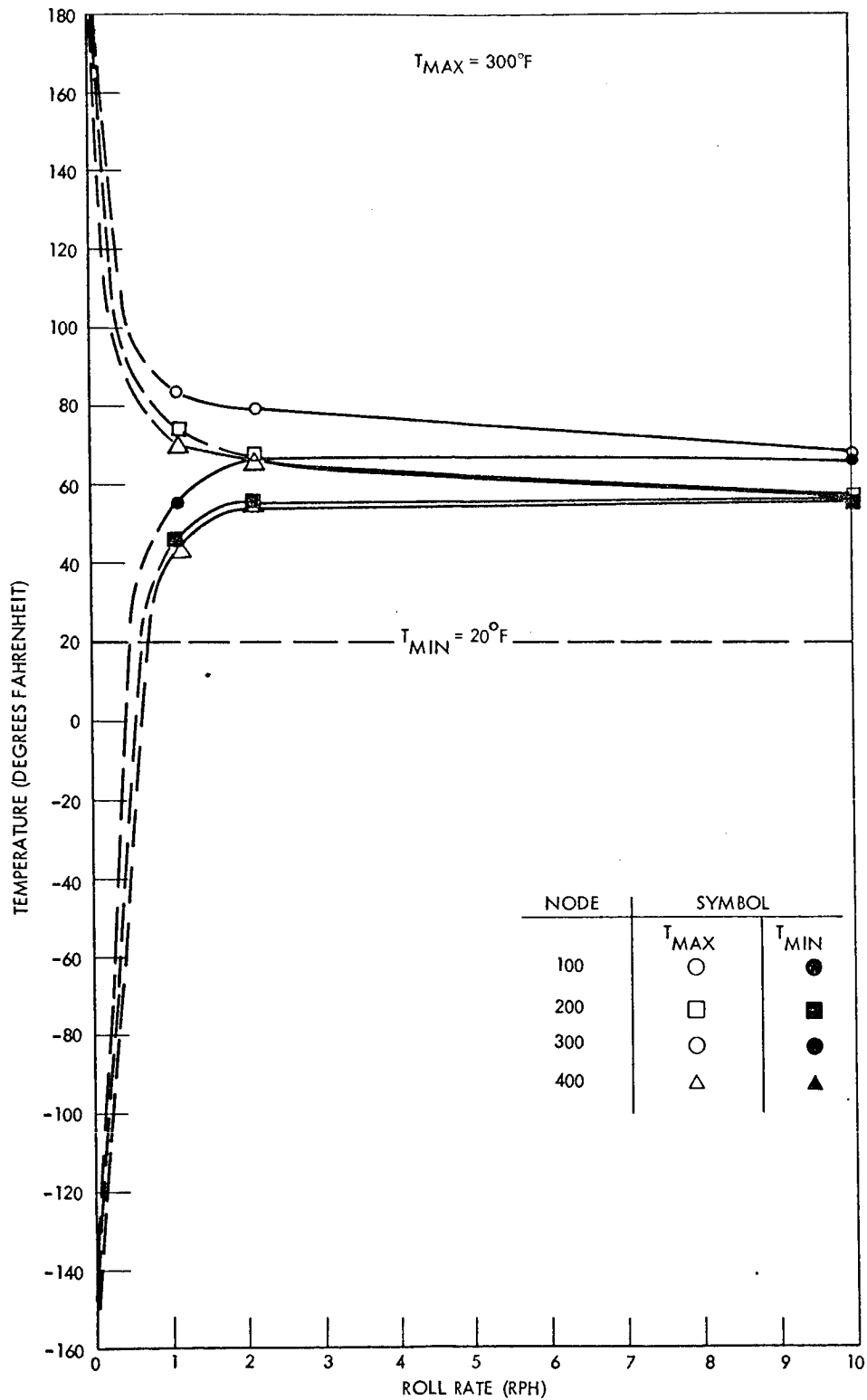
Figure 32. Maximum and Minimum Temperatures of the SM RCS Injectors During Cislunar Versus PTC Roll Rate

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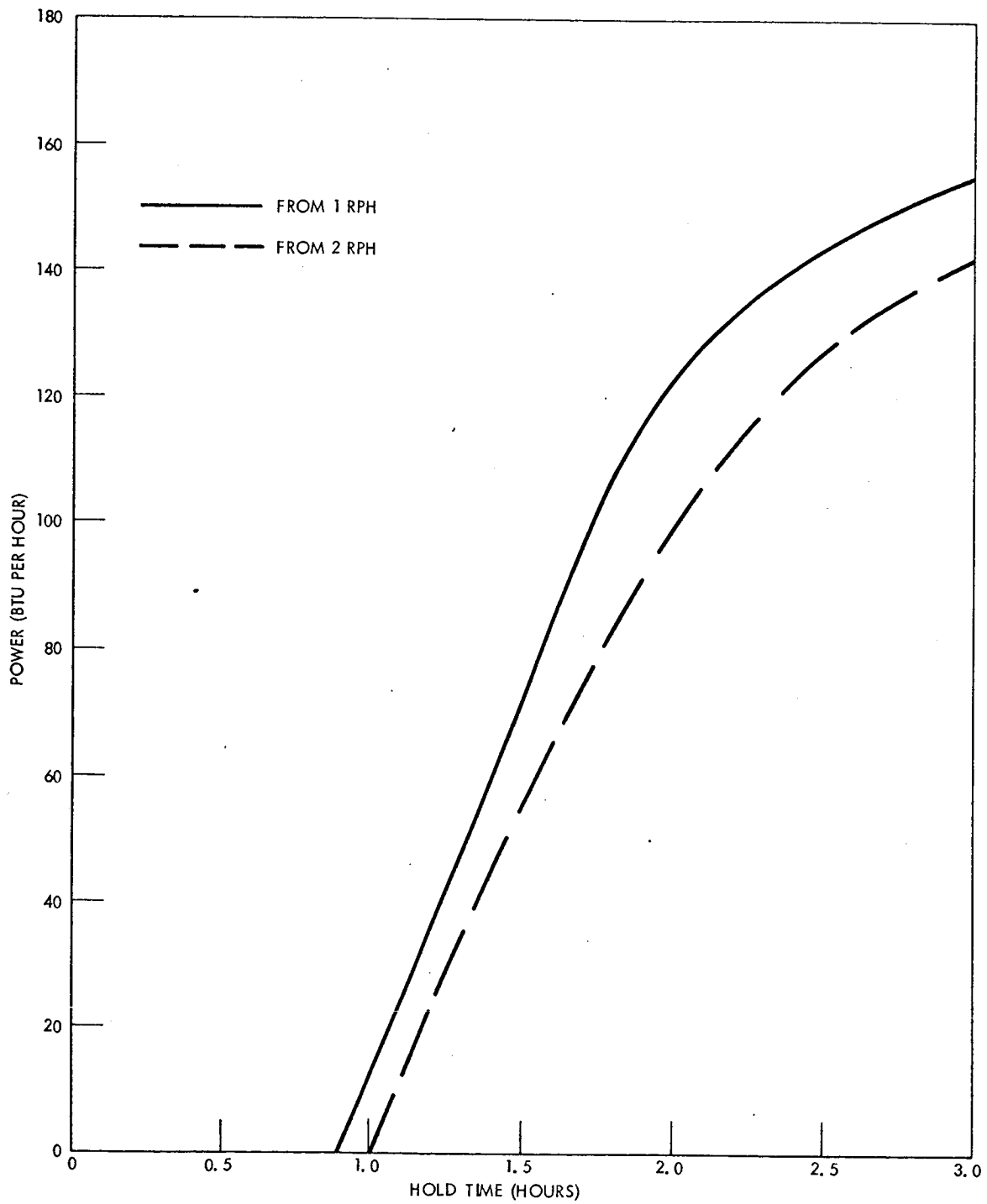
Figure 33.   Power Versus Hold Time During Cislunar
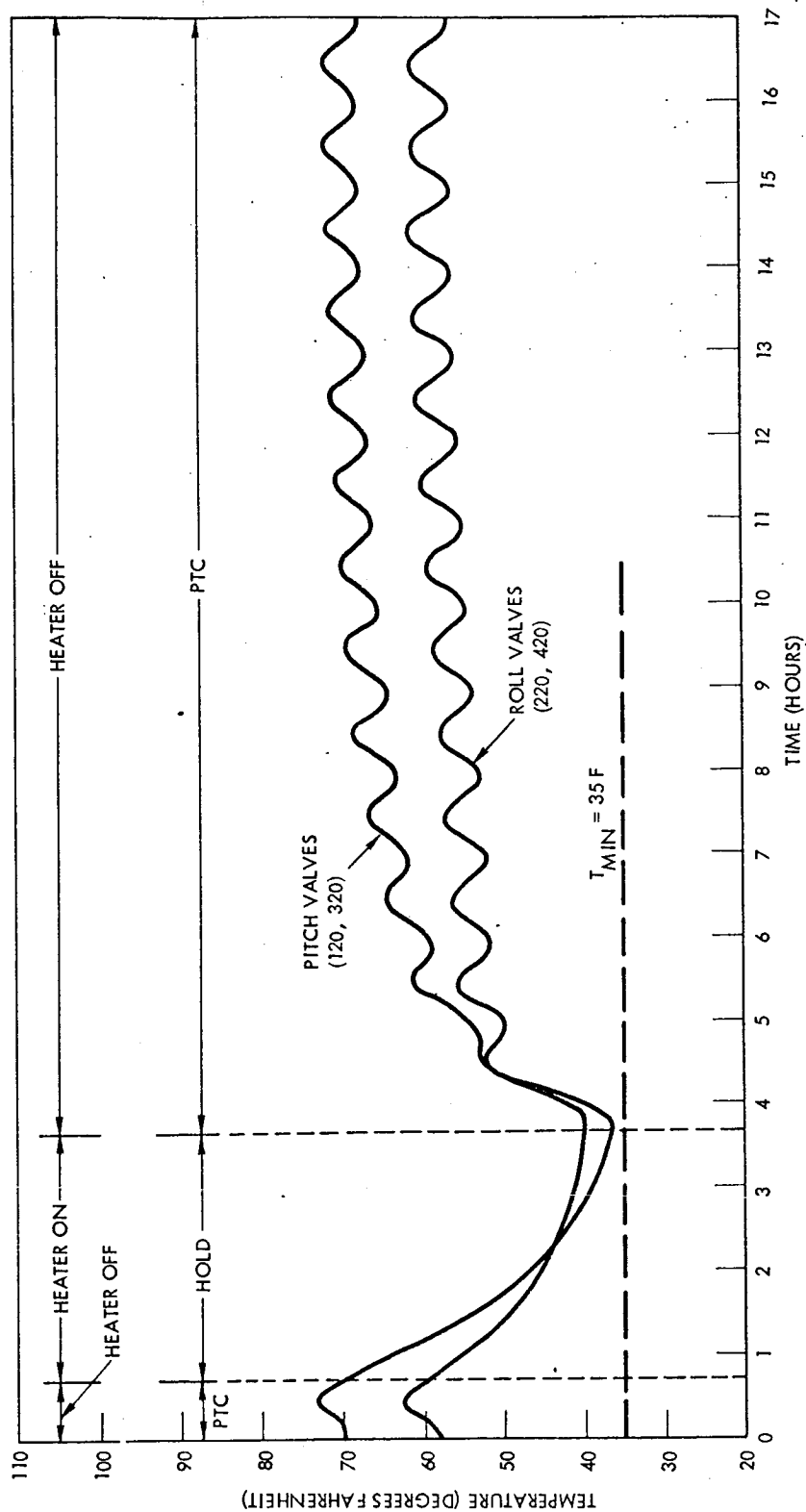
Figure 34. Temperatures of the Pitch and Roll Engine Propellant Valves During Cislunar for a 1-RPH PTC Roll Rate, Three-Hour Hold, and Recovery
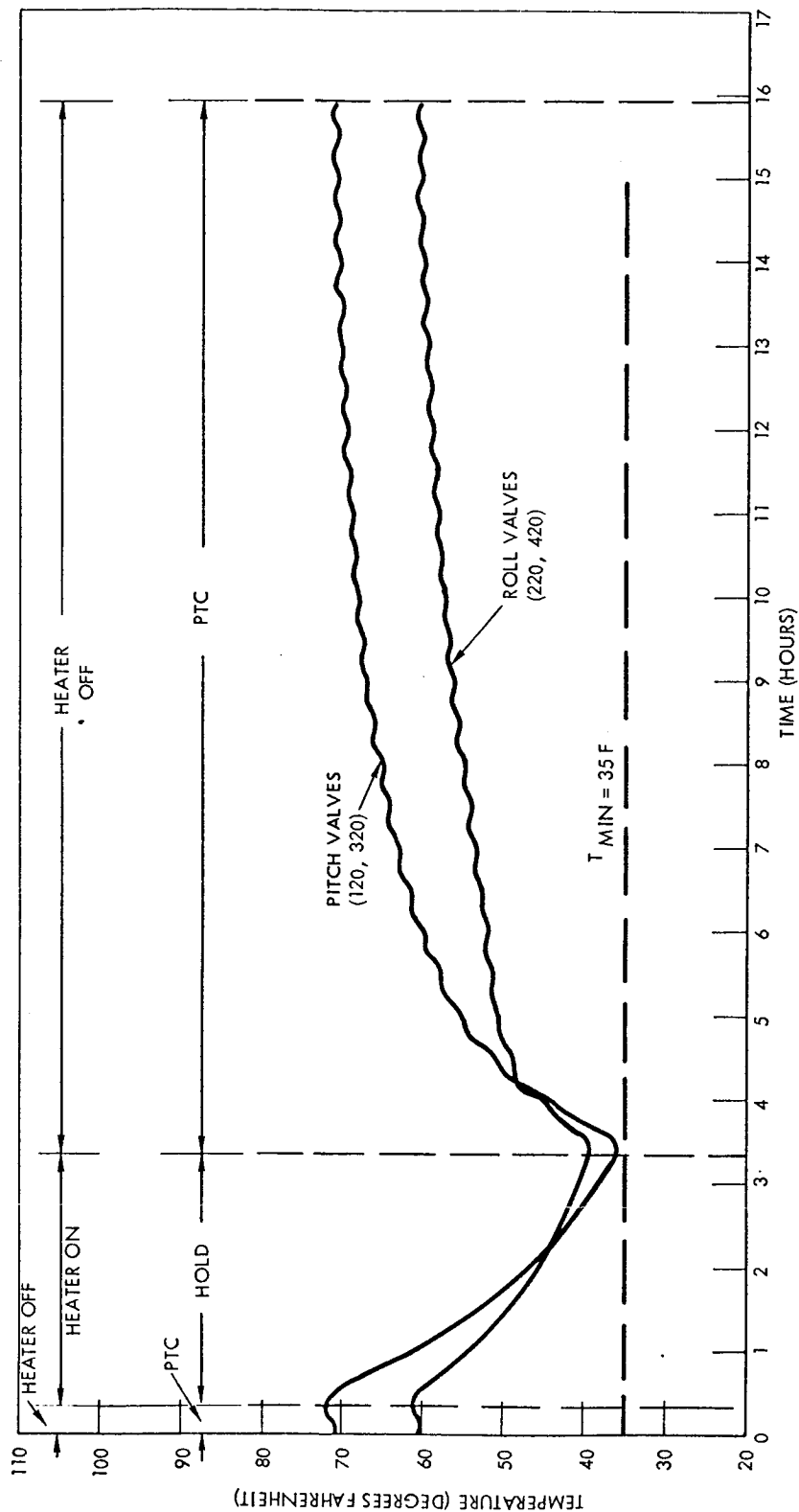
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Figure 35.   Temperatures of the Pitch and Roll Engine Propellant Valves During Cislunar for a 2-RPH PTC Roll Rate, Three-Hour Hold, and Recovery

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showed that a small increase in heater source can increase the hold time substantially beyond three hours. The maximum recovery time from a three-hour hold was determined to be approximately 12 hours.

The results of the orbital analyses indicated that, without a heat source, injectors and/or propellant valves temperatures can exceed their lower temperature limits for several cases. Further investigation in this area appears necessary.

SERVICE PROPULSION SYSTEM

A thermal analysis was performed on the SPS and adjacent SM during the orbital and transit phases of the AES missions. The objective of this analysis was to determine thermal problems which might be experienced by the system during the orbital and transit phases of the AES missions and vehicle attitude constraints that result from these thermal problems.

Components of the SPS associated with propellant flow have critical temperature limits which can not be exceeded without serious hazards. Minimum allowable temperature of the propellants in the tanks, lines, and disconnects is 40 F; maximum allowable temperature of the propellants in the tanks is 80 F, whereas the maximum allowable temperature of the propellants in the lines and disconnects is 140 F. The maximum and minimum allowable temperatures of the SPS gimbal bearings are -10 F and 200 F, respectively. During nonoperational phases, the minimum and maximum allowable temperatures of the yaw and pitch actuators are -10 F and 140 F, respectively.

The thermal model employed in the SPS analysis was a modification of an existing Block I model. Complete conversion from the Block I thermal representation to a Block II thermal model was not fully realized, e.g., the thermal description of the SM aft heat shield and close-out employed in this analysis was the same as that for Block I. Also, in the case of the propellant feed subsystem, only a partial modification of the thermal network was possible.

Seven environmental cases were investigated in this analysis: two earth orbits, four lunar orbits and one cislunar case. Vehicular orientation assumed for the cislunar analysis was with the X axis held perpendicular to the solar vector. For synchronous earth orbits, the vehicle is in the earth's shadow approximately one out of 24 hours. Hence, the environmental conditions during the cislunar case are essentially the same as those that exist for synchronous earth orbits with the X axis perpendicular to the solar vector.

The SPS thermal analysis indicated that, in general, the system is cold-biased. Thermal problems existed in all the cases investigated. However, these problems are confined to two subsystems: the propellant feed subsystem and the gimbal actuator assembly. The temperatures of the propellants in the feed lines and transfer lines were determined to be less than the allowable limit of 40 F. For most of the cases, these temperatures

were marginal. A refined thermal model might indicate there are no problems; hence, it is recommended that the propellant feed system, especially the feed and transfer lines, be investigated in detail. In three of the four lunar orbits and in one of the earth orbits, the yaw actuator temperature was less than the allowable temperature limit of -10 F.

It is recommended that a refined thermal analysis of the SPS be conducted during the orbital and transit phases of the AES missions. This may be accomplished by generating a detailed thermal network describing the complete SPS; it should reflect the AES vehicle design. In support of the refined thermal analysis, a study should be conducted to accurately define the thermal environment for each case considered.

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PARACHUTE COMPARTMENT THERMAL ANALYSIS

A thermal network analog of the Block II parachute compartment structure was used to calculate temperature histories and expected temperature extremes of compartment components with respect to the requirements of AES missions. Specifically, it was determined whether parachute and ablator heat shield materials could be maintained within tolerable limits without the addition of active temperature control devices.

Computer results for indefinite holds in black space showed that the ablative heat shield could be held above its lower temperature limit of -150 F by adjusting the insulation effectiveness of the heat shield insulation. Decreasing the effectiveness tends to raise ablator temperatures, but it also causes parachute temperatures to decline. With the heat shield at -150 F, parachute temperatures were well above the lower temperature limit (-65 F) of the compartment interior.

Determination of the maximum expected heat shield equilibrium temperature (that is, facing the sun) showed that the ablator slightly exceeded the upper limit of 150 F. This result is considered conservative because of the assumption that no heat was transferred internally to the compartment. In actuality, parachute temperatures would be within allowable limits.

The results showed that PTC management, by rotating the vehicle around the X axis, was satisfactory as a means of equalizing high and low temperatures after space hold conditions. Figures 36 and 37 illustrate the temperature history of ablator surface nodes numbers 41 and 91 during thermal cycling and hold procedures.

A "hot" lunar orbital case in which the compartment was subjected to maximum lunar emission and reflected solar radiation on one side and direct solar heating on the other did not cause ablative heat shield temperature to exceed allowable limits. Parachute temperatures remained well within temperature limits.

Drogue chute mortars and recovery aids remained within their temperature limits in all cases analyzed.

The final conclusion is that the parachute compartment temperatures can be controlled by choosing the appropriate thickness and type of insulation; thus, active devices such as heaters are unnecessary. It should be noted that an optimum insulation value was not obtained in this case.

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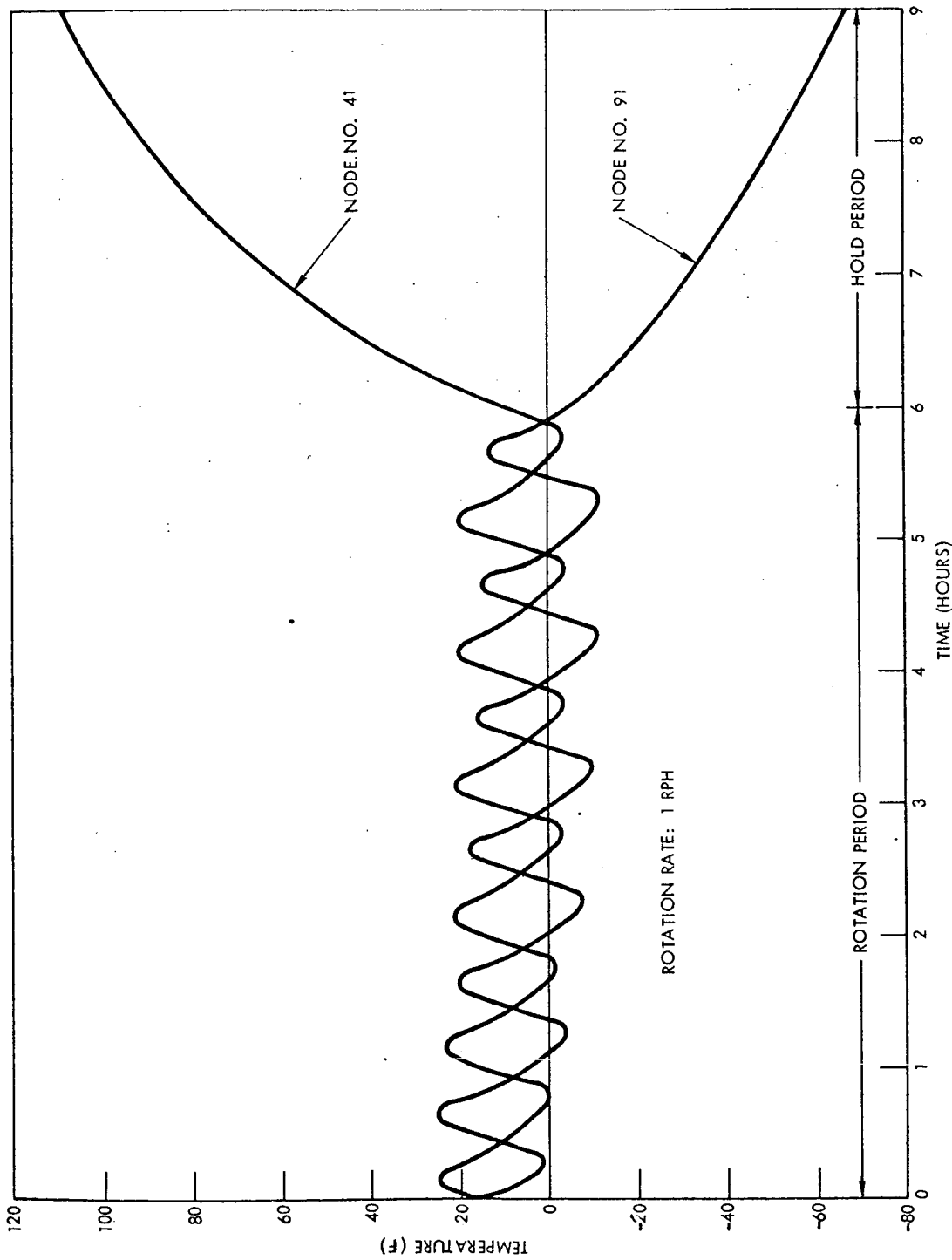
Figure 36. Temperature-Time History of Exterior Surface Nodes Nos. 41 and 91
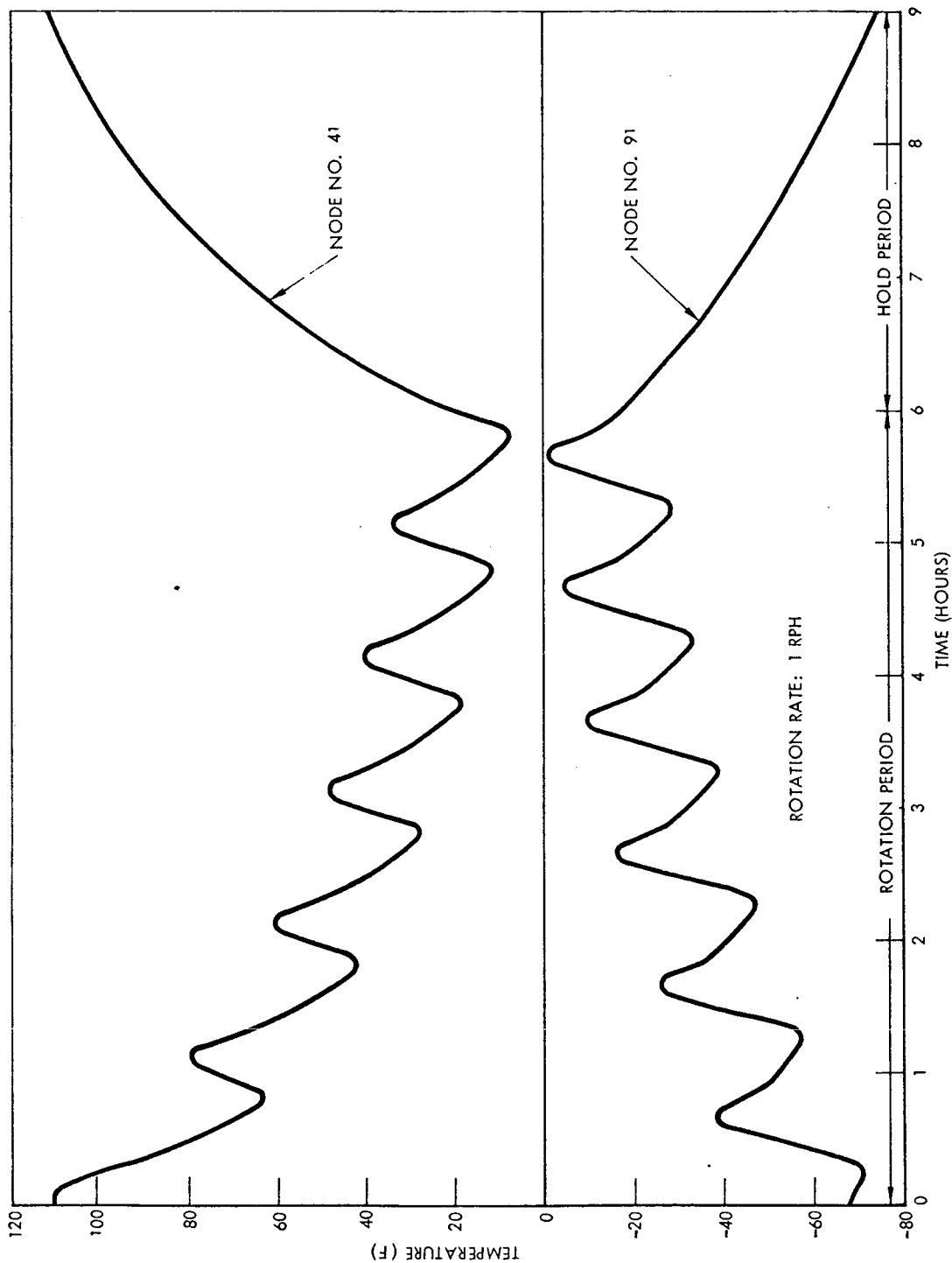
Figure 37. Temperature-Time History of Exterior Surface Nodes Nos. 41 and 91

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THERMAL ATTITUDE CONSTRAINTS

A preliminary evaluation of the attitude constraints required for the thermal control of AES flights has been made. The evaluation was obtained by extrapolation from the attitude constraints necessary to insure thermal control of the Block II hardware during the Apollo design reference mission. It is intended that this information be considered as preliminary criteria for mission profile design and budgeting of consumables for the AES flights.

Temperature and performance limits on several subsystems are responsible for the thermal control attitude constraints. The subsystems governing attitude management are the SM RCS and ECS. Passive temperature control (PTC) must be provided these subsystems. PTC is the maximum application of coatings, materials, configuration, and thermal environmental control by attitude management. It is applied as a means of maintaining the spacecraft within allowable temperature and performance limits. PTC is utilized in lieu of or in conjunction with electrical heaters. It is mainly feasible during cislunar flight phases when long periods of inactivity and constant exposure to the sun are possible. When maneuvers and operations require attitudes which do not permit PTC, heaters assist in maintaining temperatures within acceptable limits. Heaters are provided each of the four SM RCS quads so that fixed attitude holds may be maintained for extended periods. The heaters are sized to maintain the critical components of the engines within temperature limits for a maximum period of three hours without the benefit of environmental heating. The heaters are unnecessary when PTC rolling is used. Ordinarily, each heater is thermostatically controlled, but manual override operation is provided. It is most desirable to maintain the quad temperature by the use of thermal cycling whenever possible in order to economize on the electrical power budget for the RCS engine heaters.

The economical solution to thermal control is accomplished by uniformly illuminating the spacecraft. Cyclically rolling the spacecraft about the X axis at a rate of 1 to 2.5 revolutions per hour while the spacecraft X axis is held perpendicular to the sun's rays is the most feasible and economical maneuver. This technique limits CSM temperature excursions.

The ECS radiators provide for the thermal control of the ECS by rejecting heat produced by crew metabolism, electronic equipment dissipation, etc. The heat rejection rate varies over a three-fold range, depending upon environmental conditions and fuel cell energy production. Performance of the ECS radiators is dependent upon environmental heating incident upon them,

in addition to the heat rejection loading. Supplemental heating and cooling is provided to accommodate the extremes of the operational range in conjunction with unfavorable attitude. At the high end of the heat rejection range (with the radiators in a well illuminated attitude), supplemental cooling is provided by a water evaporator operating on water produced by the fuel cells. At the low end of the heat rejection range, with the radiators in the least favorable illumination attitude, supplemental heat loading is provided by a 300-watt heater to prevent the coolant from freezing completely in the radiator circuit.

Lunar orbit operation may impose excessive requirements on the radiators if the spacecraft is allowed to remain in a worst case attitude. Worst case attitude in lunar orbit occurs in the region of the subsolar point on the moon, when one radiator panel faces the sun and the other faces the moon. Excessive water boil-off results when the spacecraft is in this attitude. The water boil-off rate may be sufficiently high in lunar orbit to use up all water if repetition of the worst case attitude occurs. To enhance the radiator effectiveness, thermally unfavorable attitudes are to be avoided over the subsolar region during orbit. The consequent attitude restriction is that ECS radiator edges are to be pointed toward the moon when the space-craft is within 25 degrees of the subsolar point. Exceptions to this attitude are allowed on no more than three consecutive orbits or on no more than eight total orbits in one lunar mission. The condition of subsequent equal time in the ECS preferred attitude is required.

LOW-INCLINATION EARTH ORBIT

Low-inclination orbit is essentially equivalent thermally to the 33-degree inclination Apollo orbit. No attitude restrictions result for this mission. The RCS heaters are activated as required by the thermostats, and energy requirements are budgeted. The ECS radiators may be supplemented by water stores taken on prior to launch. It is not expected that the ECS makeup heater will be required.

LOW-INCLINATION LUNAR ORBIT

The AES low-inclination orbit is essentially equivalent thermally to the 5-degree inclination Apollo orbit. The ECS preferred attitudes over the subsolar point are required. No more than the allowable number of exceptions is permitted, and equal time in the preferred attitude is required. The RCS heaters are activated, as required, by thermostats. The AES energy budget must reflect these Apollo requirements.

SYNCHRONOUS EARTH ORBIT

Synchronous earth orbit is thermally equivalent to the cislunar mission phase of Apollo. PTC thermal rolling with the spacecraft aligned broadside with respect to the sun or the equivalent is required. Stationary attitude is permissible but requires that the RCS heaters be activated within about 50 minutes. Depending on attitude, the ECS heater may be activated immediately or not at all. RCS electrical heater power is budgeted for a mission on the basis of repeated sequences of a three-hour hold with heaters active, followed by 15 hours of recovery by PTC rolling with heaters inactive. ECS makeup heater power is budgeted on a basis of 10 hours cumulative operation. It is expected that the SPS may become critical and require certain preferred attitudes during synchronous orbit.

POLAR EARTH ORBIT

This mission is preferably treated as two types representing thermal extremes. The first type is the orbit plane near or including the subsolar point and is thermally identical to the previously discussed low-inclination earth orbit. Identical variation and intensity of environmental heating are experienced by the CSM. All previous comments apply here.

The other-extreme is the orbit plane near or coincident with the terminator plane. In this orbit, the CSM is always irradiated by sunshine and earth emission. The optical properties of the radiator coating are such that the net absorption of energy is nearly the same for sunlight as for earth emission. In certain attitudes, one radiator may partially face the sun, while the other may face the earth. This attitude, in conjunction with a high heat rejection rate, will demand water boil-off rates in excess of water production rates. Only detailed analysis will indicate whether a full mission may be completed. Other attitudes require a continuously active heater in at least one RCS quad.

POWER DISTRIBUTION AND GENERATION SYSTEMS

POWER DISTRIBUTION AND GENERATION SYSTEMS

The function of the AES power generation and distribution system is to generate and distribute all necessary power for the spacecraft. This system consists of the fuel cells and EPS radiators, batteries and associated equipment, inverters, and all equipment necessary to distribute and control both ac and dc power. The AES power system concept is basically the same as that for the Apollo Block II in that prime energy is obtained from the reaction of oxygen and hydrogen which is converted to dc electrical energy via fuel cell power plants. Ac power is obtained by conditioning and inverting the dc. Fuel cells are located in Sectors I and IV of the service module, except for a special mission case where Sector I is kept open for experimental equipment.

During normal spacecraft operation, two fuel cell powerplants operate in electrical parallel. Two other powerplants are provided for redundancy and are stored at ambient temperatures until required. Power loads above the combined output of two powerplants are supplied by supplementary batteries (located in the service module) which are automatically paralleled with the fuel cells as required. These batteries can also supply power for starting redundant fuel cells in the event of sudden fuel cell failure. Separate batteries (located in the command module) provide power for all spacecraft operations during reentry, landing, and postlanding mission phases.

Major differences from the Block II system are in the areas of component operating life, the number of fuel cells, their characteristics, the way they are used, the supplementary battery system, and provisions for supplying power to an external device. The major new components are fuel cell start-up provisions (including an automatic start programmer), a fuel cell voltage limiter, the fuel cell electrodes, and the supplementary battery system.

For a more detailed description of the power system studies, see SID 65-1525.

POWER DISTRIBUTION AND CONTROL

The AES power distribution and control subsystem obtains primary electrical energy from the fuel cells from launch to earth entry, and from

storage batteries from entry through postlanding. The power distribution system controls, conditions as necessary, and distributes this energy to all spacecraft electrical loads. The power output from any operating pair of the four fuel cell power plants is connected by power motor switches to two redundant main dc buses in the service module. The main dc buses extend to the command module where the main load distribution center is located. From this point, power feeders can supply up to 2400 watts maximum dc power to an interface connector in the command module forward tunnel.

Three entry and postlanding batteries in the command module are connected to the main dc buses to provide all power during entry and postlanding periods and may be used to supply emergency peak power during other flight periods. A battery charger in the command module recharges these batteries to replace charged stand losses and energy drained for in-flight battery bus loads.

Pyrotechnic initiator power is supplied by two redundant pyrotechnic batteries located in the command module which supply isolated redundant pyro initiators in the sequential events control system.

Two supplementary batteries located in the service module are connected to the service module main buses to supply fuel cell in-flight start heater energy and supplement fuel cells during peak load periods. Two supplementary battery chargers in the service module will provide recharge energy from the fuel cells during minimum load periods.

Three redundant inverters in the command module invert dc power to provide regulated 115/200 volt, three-phase, 400 cps ac power for nine subsystems.

Power distribution system changes from the Block II subsystem include: provisions for connecting power output of the fourth fuel cell to the service module bus, new power feeders for an external module, supplementary batteries and battery chargers, possible redesign of the pyrotechnic batteries for increased charged stand time, and possible redesign of the inverter for improved reliability.

POWER REQUIREMENTS AND SYSTEM RESPONSE ANALYSIS

In order to define and to predict the performance of an electrical power system, it is necessary to know what the load will be, both from the standpoints of energy required and the power demand as a function of time. From a power profile and the characteristics of the power sources and the distribution system, voltages and currents anywhere in the system and at any point in time can be computed.

The power loads are divided into two categories, housekeeping loads that provide only the power to sustain the spacecraft, and the total load profile for the spacecraft (including a laboratory) and the experiments. The average housekeeping loads are shown in Table 36.

Table 36. AES Housekeeping Load (Including Inverter Efficiency of 75 Percent and 4 Percent Cable Losses)

System	AC		DC	
	Peak	Average	Peak	Average
Communications	322	19	64	50
Crew Systems	0	0	6	3
Cryogenic Gas Storage	123	46	399	50
Controls and Displays	0	0	23	19
ECS	292	292	63	63
Fuel Cells	209	209	25	25
G&N	22	0	550	48
Illumination	76	66	49	47
RCS	0	0	308	37
SCS	161	0	117	0
SPS	96	0	4580	0
Instrumentation	49	49	131	131
		681 watts		473 watts

Average Housekeeping Load = 1154 watts

Of the four reference missions analyzed during the study, the lunar mapping mission (No. 3) was selected as the design mission since it represents the most difficult one from a power system standpoint. The mapping equipment imposes severe peaks and power transients on the system resulting in a total load profile as shown in Figure 38.

From a total energy standpoint, the 45-day synchronous earth orbit mission (No. 2) was the most critical and required approximately 1918 KWH. This mission however, was not used to size the cryogenic storage system; rather, the maximum quantity that could be fitted into Sectors I and IV of the service module was provided. This allows power load growth from present estimates and/or increased mission experiment capability.

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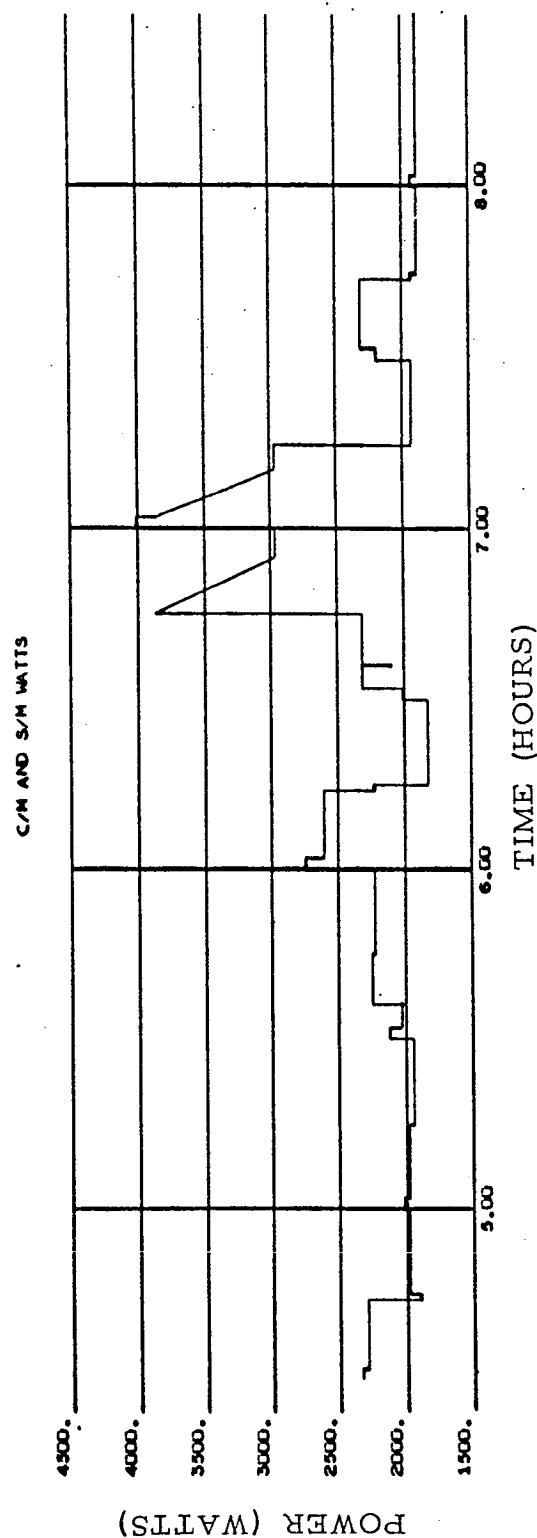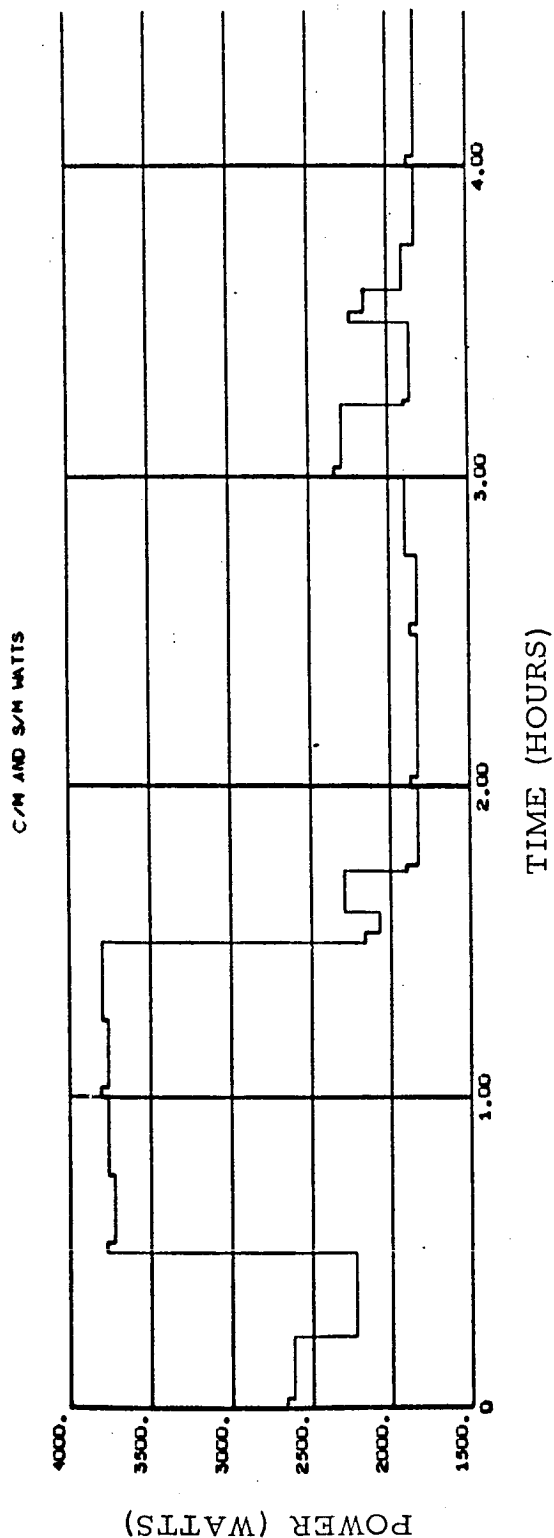
Figure 38. Load Profile, AES Mission 3, Lunar Orbit Phase

POWER DISTRIBUTION

The principal differences in AES power distribution system from Block II are in equipment operating time, duty cycles and reliability requirements, and possibly in the amount of power supplied to an external device.

During AES reference mission number 3, the lunar mapping mission, the largest power load could be the experimental module. The gage, current-carrying capacity, and weight of the wiring to carry the laboratory power were calculated and are shown in Table 37.

Table 37. Electrical Wiring to LEM Laboratory

Minimum Interface Voltage	Wire Resistance	Wire Gage	Wire Weight
26 volts	14.1 milliohms	#6	1.88 lb.
25 volts	23.0 milliohms	#8	1.19 lb.
24 volts	30.1 milliohms	#10	0.73 lb.

From calculations on wire size, resistance and voltage drop for the dc power feeder from the command module main bus to the CSM laboratory interface, it was determined that no excessive wire weight will be required to maintain the standard Apollo electrical power characteristics at this interface.

POWER CONVERSION

Fuel cell dc power is inverted to 400 cps ac power in the AES by an inverter system consisting of three redundant inverters, dual redundant ac buses, and nine load systems. Differences from Block II include the ac loads and inverter reliability requirements.

The average inverter load during CSM housekeeping operations is 511 watts. The maximum continuous ac load condition for one inverter is 1413 va without fuel-cell pump power-factor correction, and 1224 va with power-factor correction capacitors.

It was determined that the rated performance of the Block II inverter is adequate for AES ac loads, including 100 va reserve for ac loads, provided that power-factor correction is furnished for the fuel cell pump motor loads, as on Block II.

The Block II inverter will meet the Block II reliability requirements per the latest Westinghouse inverter reliability analysis. It would not meet the required MTBF for the AES in a three inverter system. Two methods of inverter system change to meet AES reliability requirements are summarized in Table 38.

Table 38. Two Methods of Inverter System Change to Meet
AES Reliability Requirements

Delta From Block II	1. Add Fourth Inverter	2. Improve Inverter Reliability
Weight Change, CM	+50 lb	-14 lb
Weight Change, SM (F/C reactants saved on 45-day mission due to lower inverter losses.)	0 ———	-42.3 ———
Total Weight CM and SM	+50	-56.3
Efficiency, full load half load ·	74%, 0.9 P.F. 72%, 0.9 P.F.	80%, 0.70 P.F. 76%, 0.90 P.F.
CM Space Required	1480 cu. in.	No change
Estimated Development Cost	No change	$740,000
Estimated Production Cost, per CM	$45,000	No change
Reliability for Crew Safety, Inverter System	0.999995	0.999991

Improved inverter reliability is recommended over adding a fourth inverter. Methods of improving inverter reliability are: (1) inverter circuit design changes including circuit simplification, operation of components at lower ambient and electrical levels, and replacement of components with more reliable types; and (2) component reliability improvement program including selected critical components, and reliability testing of and design changes in components.

PYROTECHNIC BATTERIES

For the AES missions, the pyrotechnic loads will be essentially the same as on Block II. This being the case, it would seem feasible to utilize

the same battery design for the AES missions. This, however, is not the case. The present batteries would not be able to meet the extended activated wet-stand time of the AES missions.

The pyrotechnic battery specification requires that the capacity of the battery shall not be less than 45 ampere-minutes after a 36-day wet-stand when discharged at 75 amperes for 36 seconds at ambient temperatures ranging from +50 F to +110 F. For a 45-day mission, a 60-day wet-stand time will be required. Therefore, it is necessary to modify these batteries to increase the wet-stand time. This can be done by adding two additional layers of cellophane per plate, with the result that an additional 1.44 inches in length and 175 grams of additional weight would be required for the 20-cell battery.

POWER FOR PEAK LOADS

Of the four reference missions, only the 34-day lunar orbital mission requires, at present, peak power for relatively long periods of time if the presently defined mission loads are assumed. Upon establishment of lunar orbit, peak loads will continue to cycle for a cumulative 28 days. The repetitive cycle will consist of (see Figure 38) 3800 watts average for 1 hour, 5.25 hours available for recharge; 3500 watts average for 1/2 hour, 5.25 hours available for recharge. This cycle will continue for the 28-day orbital mode of operation, thereby imposing 112 cycles of battery operation. The parallel combination of two fuel cell modules will handle approximately 2550 watts, thereby requiring 1250 watts from the supplementary batteries.

Two methods to supply the peak experimental load demands of the 34-day lunar orbital mission were investigated. The first consists of continuous operation of three fuel cell modules, and the second operates two modules together with a supplementary battery pack.

The power required for the fuel cell parasitic loads and supplementary battery-charge losses were included in total KW-H fuel cell output and reactant consumption calculations by the EPS system response analysis program for one typical 24-hour period in lunar polar orbit. Results are shown below:

Methods of Supplying Peak Load	Energy for 24-Hour Period	
	KW-H	Pounds Reactants
3 fuel cells	65.50	50.24
2 fuel cells, supplementary battery	50.83	39.62
Difference, in favor of supplementary batteries		10.62
		x 28 days
		298 pounds of reactants

The system-response computer program showed that, even with three fuel cells operating, the voltages at the load interface were out of tolerance for a short time after the load transients. Therefore, supplementary batteries were chosen to supply peak loads.

SUPPLEMENTARY BATTERIES FOR PEAK LOAD AND FUEL CELL START

Either silver-zinc primary or secondary batteries can be considered for in-space fuel cell start-up. Using silver-zinc primary batteries, the total weights and volumes will be approximately as follows:

1. Command module 400-hour version fuel cell battery pack weight (2), 290 pounds; total battery back volume 2660 cubic inches. 1200-hour version fuel cells, 145 pounds; battery volume, 1330 cubic inches.

2. Service module 400-hour version fuel cell total battery pack weight, 330 pounds; total battery pack volume will be approximately 1458 cubic inches. 1200-hour version fuel cell total battery pack weight, 158 pounds; total battery pack volume, 1330 cubic inches.

For silver-zinc secondary batteries, the normal peak load to be furnished by the batteries during the cyclic period will be 1500 watts for an energy required of 1500 WH and a capacity requirement of 54.5 AH. At a prescribed 25-percent limited depth, the required battery pack capacity will be 218 AH. The required recharge per cycle period will be 60 AH at a maximum C/20 recharge rate. Three battery chargers would be required in order to charge to full capacity within 6 hours. Each charger would be rated an average 4-ampere output and weigh approximately 4 pounds.

During the restart period, the batteries would be discharged to 57.5 percent of their rated capacity. It will require approximately 14 hours to recharge the battery to maximum capacity at C/20 rate. The weight and volume required for the silver-zinc secondary battery pack would be 192 pounds and 20.52 cubic inches.

Silver-cadmium batteries could be used in conjunction with a configuration in which the CSM supplies a laboratory. The batteries would be used to handle both the peak loads plus the in-space start-up of fuel cells. If the batteries were to be used for both in-space start-up and peak loading, the capacity of the batteries would need to be sized for in-space start (larger requirement). The capacity required for in-space start would be 138 AH, or two 70 AH batteries. During the cyclic period, the batteries would be discharged to 37 percent depth of discharge.

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The battery pack weight and volume would be approximately the same for either case; 178 pounds, and 1800 cubic inches. This is the lightest configuration supplying power for in-space start of fuel cells and peak loads for reference mission 3. Therefore, for the supplementary battery-system requirements of fuel cell in-space start-up and peak loads established during this study, the supplementary battery system selected is: two 70-ampere-hour silver-cadmium secondary batteries and two chargers. The selection was made on the basis of weight, reliability, and state of development. The batteries would be located in the service module. No space for batteries of this size is available in the command module without major equipment relocation.

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POWER GENERATION SYSTEM

Of the four reference missions considered, two missions were considered to be extreme for selection and preliminary design of the fuel cell powerplant (FCP) system. These were: Mission 2, which is a 45-day earth orbital mission that includes the most stringent requirements for maximum life and mission reliability by virtue of its 1080-hour duration, and Mission 3, which is a 34-day lunar mapping mission that sets high requirements for crew safety because of the distance from the earth involved in an abort.

All of the FCP configurations in this study made use of the basic Apollo FCP design. Every one was a natural extension of the technologies which have been established by the Apollo developmental program. Minimum change thus became a basic ground rule for FCP configuration selection.

Much of the total system operating flexibility has been designed into the overall power generating system. A consideration for the fuell cell powerplants was to establish a reasonable maximum power requirement. The total system peak power runs as high as 8000 watts for short periods of time. While it is not practical to design an FCP to sustain these peaks, the power generated by the fuel cell should pick up a reasonable portion so as to minimize the weight of peak batteries. The lowest maximum fuel cell power level considered was 2840 watts, or a system with each of two FCP's operating at 1420 watts, which is the maximum continuous power rating of the Apollo FCP. The study investigated individual FCP ratings up to 4000 watts. Increases in rating had to be evaluated against other program criteria given. Technologies required and associated program costs and schedules had to be justifiable. A maximum power of 2000 watts per FCP was considered to be reasonably attainable within the context of the program.

FUEL CELL IMPROVEMENTS FOR AES

Examination of the FCP component performance and functions have pointed out three areas of performance where modification to the FCP will yield substantial increases in system capability and effectiveness. The three areas of improvement are: (1) in-flight start, (2) improved life, and (3) (3) increased power range.

In-Flight Start

An in-flight start is accomplished when an FCP installed in the space-craft and stored in a non-operating condition is brought to an operating condition at some time during the mission. Trade-off studies were conducted to determine FCP storage conditions, startup and shutdown procedures, and startup energy sources.

The cold standby storage condition is recommended for the AES power-plants. Maintaining the powerplants at warm standby temperatures of 325 F or below increases the failure rate of the powerplants over the failure rate of zero for powerplants stored in the cold condition. A weight penalty of 0.156 pound per hour of warm storage, attributed to reactants and tankage, would also be incurred. The cold standby powerplants will be pressurized on reactants at the low delta pressures, but no degradation will occur within the cell due to the electrolyte being in the solid state.

Three startup and shutdown procedures were investigated. The three procedures were manual startup and shutdown, automatic startup with manual shutdown and automatic startup and shutdown. The automatic startup with manual shutdown is recommended. An automatic start programmer (ASP) will assure the proper time sequencing of the start procedure. The ASP will have provisions for manual override of in-flight startup and for a manual electrolyte-conditioned prelaunch shutdown. In-flight shutdown would be manual, but would not provide for conditioning of the electrolyte; therefore, restart is not possible.

The heat-up system trade-off study revealed that two of three systems studied are feasible for AES missions. The heat-up energy can be supplied by either operating powerplants or spacecraft batteries. The catalytic reactor heat-up system was eliminated because extensive development and qualification programs would be required to produce space hardware compatible with the powerplants. The use of reactors would also increase the number of instruments and controls required in the CM. The spacecraft Ag-Cd batteries used for peak loads can be increased in capacity to furnish startup energy as a backup to dc bus power with little increase in actual battery weight. The only weight penalty incurred by using the batteries for heatup power would be the reactants required to recharge the batteries. This weight of approximately 7.5 pounds per start would be less than the weight of a reactor startup. It is recommended that powerplant dc bus power be utilized to heatup the standby powerplants with batteries in parallel. The batteries will also have the capability of heating the powerplants without the aid of powerplant power. The heatup time regardless of the electrical energy source is limited to 2.8 hours for 28-volt power due to the current carrying capability of the intercell heater connectors. The 2.8-hour heatup consists

of 4 kw-hrs of powerplant or battery power plus in-line heater operation from 300 F to operating temperature.

Improved Life

To meet the AES mission duration and reliability requirements with the current PC3A-2 FCP, a six-FCP configuration, two operating with four standbys, is required. Increasing the FCP life rating to 1200 hours would allow a reduction in the number of redundant FCP's to two or one, depending on the degree of development. Changes which would increase the life and/or performance characteristics of the PC3A-2 FCP are: (1) ceria coated anode, (2) electrode activation, (3) increased cell area, (4) ceria fill electrolyte, and (5) cell process refinement.

The endurance limiting process in the PC3A-2 cell has been the oxidation of the nickel cathode in the presence of a strong KOH electrolyte and subsequent reduction of dissolved nickel compounds to nickel at the anode and deposition of the anode. Because of the particular PC3A-2 cell geometry, these deposits form preferentially at certain locations within the cell, causing a dendritic type growth and a subsequent shortcircuiting of the cell. This process is self-perpetuating; as the nickel oxide goes into solution in the KOH at the cathode, it is removed from solution and deposited at the anode. It has been found, however, that if a diffusion barrier is placed between the anode and cathode, nickel deposition on the anode is eliminated. This diffusion barrier is a coating of ceria-chromia mixture applied to the diaphragm and sinter of the hydrogen electrode assembly by plasma spraying. This approach provides a minimum cost and a minimum time method for extending the endurance of the PC3A-2 FCP. A small performance penalty associated with the coating may be compensated for by a small adjustment in operating temperature.

Results of single cell, multicell and FCP testing has indicated that the corrosion process on the oxygen electrode is highly dependent on operating temperature. Reduction in the cell operating temperature results in a significant increase in cell life. A cell process change that will allow lower temperature operation with no loss in performance is electrode activation. Activation can be accomplished with only minor process changes to the PC3A-2 electrodes.

Another method of increasing the endurance of the PC3A-2 is to increase the area of the electrodes. The larger cell area will allow operation at a lower temperature while still meeting the power and voltage requirements. This can be accomplished with no penalty in reactant weight.

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By filling the electrolyte cavity with a ceria powder-KOH matrix, the endurance expectancy of the PC3A-2 cell can be increased. Experimental evidence at P&W and shown below indicates that with this type of cell, endurance, even at an operating temperature as high as 450 F, is well in excess of AES requirements.

85% KOH; 450 F, 25 psia)				
Load time, hours	0	1000	2000	3000
Open circuit voltage	1.13	1.12	1.13	1.13

In the area of electrode manufacturing, improved processing and quality control will result in more uniform sinters, hence increasing the cell reliability by reducing the level of random failure rate. Uniformity in sinter structure also results in increased repeatability of cell performance. Cell process refinement would have the same benefit for all approaches, that is, increasing the effective cell life.

Increased Power Range

The required maximum normal power of the AES FCP is identical to the PC3A-2 FCP at 1420 watts. However, the study ground rules indicated that a higher power level would be desirable, as auxiliary battery requirements would be minimized.

Increased power levels can be obtained from the PC3A-2 FCP through modifications to the power sections, such as: (1) electrode activation, (2) increased cell area, (3) ceria fill (to permit increased module operating temperature), (4) voltage limiter, and (5) primary regenerator bypass valve.

One of the methods studies to assure meeting the AES mission objectives employed an electrode-activation process. The activation process permits a trade-off of higher cell performance against cell operating temperature, as discussed earlier. With an increased voltage at a given current density, the powerplant is capable of producing more power within the same voltage limits. The power range is increased because the voltage at high current densities is increased significantly more than at low current densities.

The primary advantage of a large cell area is the increased power capability at no increase in powerplant operating temperature. Correspondingly, as the maximum power capability is increased, the minimum self-sustaining power level is increased, since overall performance is higher at all power levels. However, the increase in minimum power is much less than the increase in maximum power capability.

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Ceria fill is used to provide an increased power range through increased operating temperatures with no loss in the initial powerplant life span. The increased operating temperature enables the powerplant to reach a higher maximum power within voltage regulation. Even though the overall perform-ance is higher, the increase in maximum power is much greater than the corresponding increase in minimum power; hence, an increased power range is realized. The ceria matrix design also allows removal of the present cell diaphragms and electrode ribs. This results in a powerplant weight reduc-tion and in an increased effective cell area. This increased area adds to the power range, as previously discussed.

Because of improvements in performance or increases in power range, six of the eleven configurations presented in this report utilize a voltage limiter to limit the bus voltage at low power levels to the required 31 volts.

The voltage limiter allows operation of configurations with improved performance, since low power voltage regulation constraints are eliminated. Shifting the operating line increased the power range of the FCP because the maximum power increases while the minimum bus power remains the same. This is illustrated in Figure 39. The only system constraint is an upper limit on FCP temperature at average power, as dictated by lifetime consider-ations. The increased operating line yields significant increases in voltage at all powers. Hence, reactant and tank weight savings occur at all power levels above the limiter "on" point.
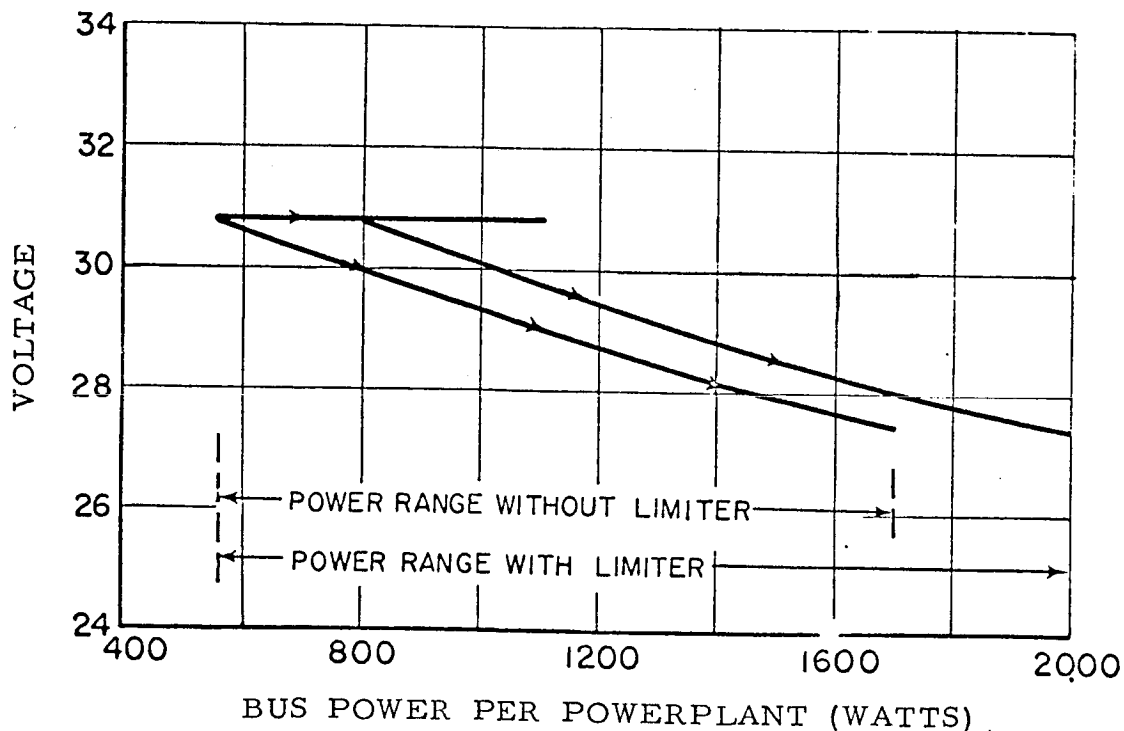
Figure 39. Effect of Voltage Limiter Operation on Power Range

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Improvements in performance and power range may require adjustment to the primary regenerator bypass valve. The PC3A-2 II valve incorporates a redundant liquid-filled bellows to provide the actuating force. This type of actuator is employed on all FCP configurations. The crack point of the valve is adjusted, however, to meet the requirements of each configuration. One configuration incorporated an additional modification of the PC3A-2 valve to allow temperature reset during operation. Depending on whether two or three FCP's are operating in that FCP system, the valve selects one of the two positions of the regenerator bypass reset mechanism. When two-FCP operation is called for, the reference temperature for valve opening is increased approximately 30 degrees. Both FCP's then operate at the increased temperature level in order to maintain the total system power capability. Hence, the power range with a reset-type valve extends from minimum power per FCP when three FCP's are operating to maximum power per FCP when two FCP's are operating.

The choice of a bypass-valve temperature schedule affects the power range and transient capability of an FCP, since it determines the operating temperature at any power level. An increased range in power is obtained as the operating temperature level is increased. A valve which requires a large FCP temperature change to effect a change in setting will yield a larger power range when compared to a valve which requires a small temperature change. This is illustrated in Figure 40. The large ΔT valve has improved transient capability for a step increase in power, but a lesser capability for a step decrease. All configurations presented incorporate valve schedules which adequately meet the power and transient requirements imposed on them.

FCP SUBSYSTEM CONFIGURATIONS

The Pratt & Whitney Aircraft Corporation (P&W) FCP was considered in various subsystem configurations which would satisfy the objectives and requirements of the AES missions. The major basic FCP system configurations compared were:

1. The present Apollo PC3A-2 400-hour FCP with two operating units and four stand-by units having in-space start capabilities.

2. A modified PC3A FCP to provide 1200-hour service life. This configuration would utilize two operating units and two stand-by units with in-space start-up capability.

Modifications to these two basic configurations were studied in depth. In addition, configurations representing a more advanced technological development of the P&W FCP were considered. Table 39 identifies the

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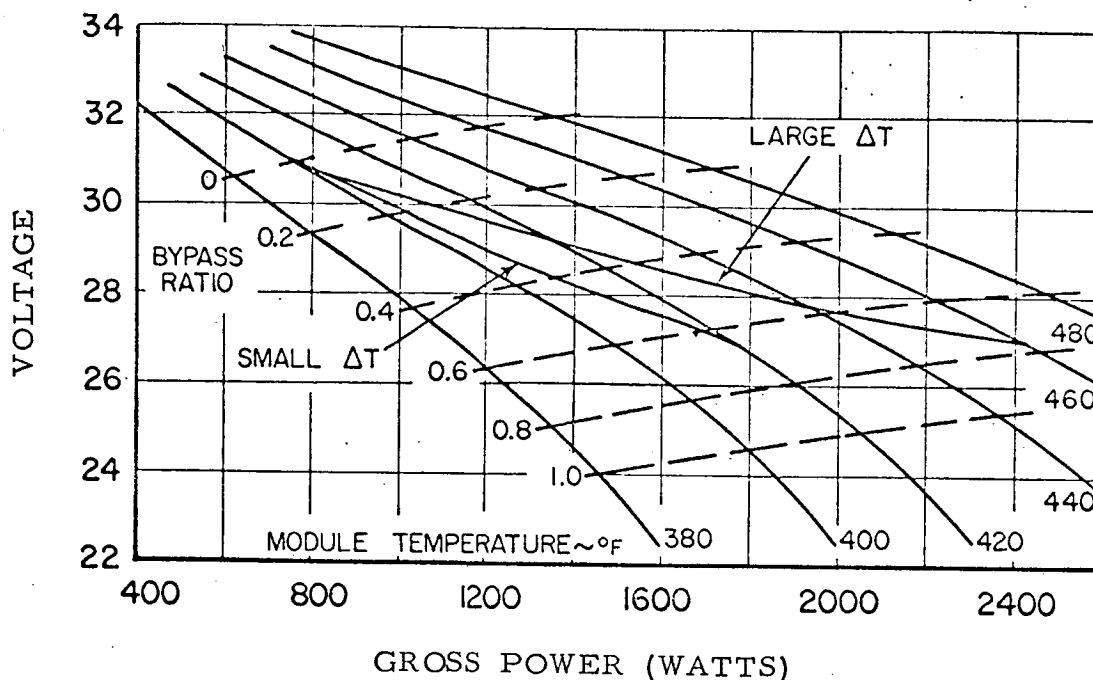
Figure 40. Effect of Primary Regenerator Bypass Valve
Temperature Setting on Power Range

various configurations investigated, and indicates the modifications required
to the FCP and/or to the subsystem concept.

Of the eighteen FCP subsystem configurations which were analyzed,
seven were eliminated after preliminary data indicated that they were
marginal in satisfying the objectives of the AES missions. The remaining
eleven were evaluated in greater depth and will be presented in the summary
trade-off table. From these 11, three were eventually selected for analysis
in depth. Descriptions of the eleven configurations follow and are identified
as approaches I through IV as shown in Table 39.

Approach I

Approach I represents the least change in the FCP configuration. Two
PC3A-2 FCP's are required to meet the mission maximum power; two
additional FCP's are required to provide 1080 hours of operation since a
single FCP has an operational life of 500+ hours. Two additional redundant
FCP's are needed to provide a total system reliability greater than 0.995.

The difference between configuration IC and configuration ID lies in
the standby mode of the spare FCP's. Configuration ID was analyzed with
the spare FCP's in the warm standby mode (325 F) to determine the effect

Table 39. Candidate FCP

APPROACH	I				II						III					IV		
CONFIGURATION	A	B	C	D	A	B	C	D	E	F	A	B	C	D	E	A	B	C
SYSTEM																		
Operating	2	2	2	2	2	3	3	2	2	3	3	3	2	2	2	2	2	2
Standby	4	4	4	4	2	0	2	2	1	0	0	1	0	1	0	1	0	0
Standby Condition	C	H	C	H	H	-	C	C	C	-	-	H	-	C	-	C	-	-
POWERPLANT																		
Changes to PC3A-2, Block II																		
Ceria Coat	-	-	-	-	x	x	x	x	x	x	x	-	x	-	x	-	-	-
Ceria Fill	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	x	x	x
Activation	-	-	-	-	-	x	-	x	-	x	x	x	x	x	x	x	x	x
Increased Cell Sinter Area	-	-	-	-	-	-	-	-	-	-	-	-	x	x	x	-	x	x
Cell Process Refinement	-	-	-	-	x	x	x	x	x	x	x	x	x	x	x	x	x	x
Number of Cells	-	-	-	-	29	-	-	-	-	29	28	28	28	29	28	-	28	26
Voltage Limiter	-	-	-	-	-	x	-	x	x	x	-	-	-	-	x	x	x	-
Two-Step Regulator	x	-	x	x	-	-	x	x	-	-	-	x	-	x	-	x	-	-
A.S.P.	x	x	x	x	x	-	x	x	x	-	-	-	-	x	-	x	-	-
28 Volt Heater	-	-	x	x	-	-	x	x	x	-	-	-	-	x	-	x	-	-
Preliminary Regen. Bypass Valve	-	-	-	-	-	-	-	-	-	-	x	-	-	-	-	-	-	-
Reset Capability	-	-	-	-	-	-	-	-	-	x	x	-	-	-	-	-	-	-

KEY: C: Cold   x: Applicable
       H: Hot   -: Not Applicable

on startup time and total system weight. If the lower recovery time is desired, a reactant and tank weight increase of 270 pounds results. No startup energy weight is included for the cold standby case.

Approach II

The configurations of this approach encompass minimum improvements in the PC3A-2 FCP required to achieve an FCP lifetime in excess of the AES mission duration. All configurations of this approach make use of a ceria sprayed anode in order to provide a 1200-plus-hour-life capability. Since two FCP's can supply the full load for the full mission duration, a maximum of two spare FCP's is required to meet system reliability requirements. To provide a 2000-watt capability per FCP, configurations IID through IIF use an improved electrode activation. Operating temperatures for these configurations are lowered 20 F to 40 F as a result of the activation. These lower temperatures provide additional margin in life characteristics as well as lowering the random failure rate. The operating characteristics of configuration IIC, are identical to those of approach IC. Total system weights and volumes are reduced since the spare FCP requirement has been reduced.

In configuration IID, the relatively high maximum power coupled with the improved performance create a requirement for voltage regulation below 800 watts. Bringing the voltage limiter on at some higher power, by changing the bypass valve crack point, increases reactant consumption at the average power of 1600 watts; bringing it on at a lower level (by dropping to a lower operating temperature line and holding maximum power operating point constant) increases reactant consumption at all power levels except maximum power.

It is possible to operate three FCP's in a load-sharing mode, as in the present Apollo system, as shown in Configuration IIF. This configuration requires a reduction in the number of cells per FCP from 31 to 29 in order to maintain voltage regulation at low power levels. A further reduction in the number of cells would increase operating temperatures to undesirable levels. In addition to requiring a reduction in the number of cells, this configuration requires a primary regeneration bypass valve with a reset capability. In the event of a failure when three FCP's are operating, the crack point of the bypass valve must be adjusted or reset upwards approximately 30 F so that the two remaining FCP's can produce 4 kilowatts.

Approach III

An additional way of obtaining at least a 4-kilowatt maximum system power capability is to increase the active electrode area which is a more significant FCP modification. Both of the Approach III configurations require two operating FCP's to provide the 4-kilowatt maximum power.

In configuration IIID, this is accomplished by a combination of increased area and electrode activation. Ceria coating is not required because the increased area allows the operating temperature to be reduced to the point where corrosion is not a factor. The increase in area requires that the number of cells be reduced to 29 in order that reasonable voltages be obtained at low levels. With 29 cells, approximately 60 watts of heater power are required to obtain a bus power of 1400 watts since the minimum self-sustaining power level is only 1600 watts. A lowering of the minimum self-sustaining power level would drop the operating temperature of the FCP to an undesirable degree and increase reactant consumption at all power levels below maximum power. An increase in the minimum self-sustaining power would create a requirement for heater power at the average power level of 1600 watts.

In configuration IIIE, the electrode area is increased to the maximum allowed by the present stack design. This additional increase permits a further reduction in the number of cells per powerplant to 28. This decrease in cells does not provide sufficient voltage regulation at power levels below 820 watts per FCP, since the maximum power capability per FCP has been increased to 2840 watts. A voltage limiter is employed to furnish the necessary regulation. A decrease to 27 cells would raise the operating temperature to undesirable levels. Configuration IIIE makes use of a ceria coated anode, which permits increased operating temperatures and therefore increased power. No reset capability is incorporated with this approach. In the event of an FCP failure, the system maximum power capability is reduced from 5680 to 2840 watts.

Approach IV

Long life, increased power, and high reliability may all be obtained with an attendant reduction in FCP weight through use of a ceria-fill diffusion barrier instead of a ceria coating. All configurations of Approach IV employ the ceria-fill cell design in addition to improvements discussed in the preceding sections.

The individual FCP configurations in configurations IVA and IVB are similar to configuration IID and IIIE respectively. Use of the ceria-fill design lowered the FCP weight with no loss in FCP life or power capability.

Configuration IVC shows the results of exploiting all improvements to realistic limits in order to obtain a 4-kilowatt, lightweight FCP with a 1200-plus-hour lifetime and high reliability. Each 26-cell FCP is capable of meeting the power range from 700 to 4000 watts by means of the increased electrode area with improved performance, while not exceeding an FCP weight of 167 pounds. The current Apollo FCP weight is 225 pounds. The

use of the advanced performance eliminated the need for a voltage limiter or valve reset capability, while the ceria-fill cell design insured adequate FCP life and low FCP weight.

FCP Weight and Volume Comparison

Table 40 itemizes the individual FCP weights. Except for the advanced type IV FCP, the unit weights do not change appreciably; hence, the total system fixed weight is mainly a function of the number of FCP utilized. The total weight and volume reduction obtained by using a type II (2 operating, 2 standby) 1200-hour FCP system as compared to a type I (2 operating, 4 standby) 400-hour FCP system is approximately 565 pounds and 19.3 ft^3. Practically all of the reduction is accounted for in the elimination of two FCP modules.

Table 40. AES FCP Weights

Configuration	Individual FCP Weight (pounds)
IC	225.4
ID	223.8
IIC	225.4
IID	228.4
IIE	228.4
IIF	219.1
IIID	247.1
IIE	247.1
IVA	198.4
IVB	224.1
IVC	167.2

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HEAT REJECTION AND RADIATOR CONCEPTS

The AES reference missions and the selection of a recommended four-FCP (2 operating, 2 cold stand-by) system improved new constraints in the Apollo Block II FCP heat rejection radiator.

For each of the proposed missions, an environmental heat input was obtained for the complete orbit. The area requirement for the radiator was divided into eight sections, and incident energy was calculated from direct solar, planetary emission, and reflected solar energy. Heat rejection requirements for various FCP configurations were considered, final consideration being given to approach IID as shown in Table 41. The values given for Approach IA correspond very closely to that of Apollo Block II and can be used as comparative values.

The present 40-ft^2 radiator on the service module fairing utilizes all of the readily available area at this location. Total area required for maximum FCP flexibility exceeds 40 square feet by at least 14.4 square feet. Therefore, it is desirable to include an auxiliary radiator located on the service module. This radiator can be located in Sector IV on the bottom of the panel. Its performance will not affect crew safety. The auxiliary radiator will be a removable panel so as not to break the primary fuel cell fluid loop. By locating the radiator on the bottom of Sector IV it is possible to have access to the SPS engine and cryogenic tanks without breaking the auxiliary loop.

Table 41. Estimated Heat Rejection Requirements

80° F (ENVIRONMENT)

FCP Configuration IA (400 HRS)   and   II C (1200 Hrs)

Range Power (w)	563	700	800	1000	1200	1420	1600	2000
Ht. rejection $\left(\dfrac{Btu}{Hr}\right)$ to Radiator	1050	1250	1450	1875	2400	3075	3675	5900
FCP Configuration IB								
Range Power (w)	563	700	800	1000	1200	1420	1600	2000
Ht. rejection $\left(\dfrac{Btu}{Hr}\right)$ to Radiator	1775	1025	1225	1600	2000	2475	2875	3850
FCP Configuration IID								
Range Power (w)	563	700	800	1000	1200	1420	1600	2000
Ht. rejection $\left(\dfrac{Btu}{Hr}\right)$	1275	1275	1275	1500	1950	2500	3050	4250

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New components will be added to the present heat rejection design to enact heat transfer from the water glycol to the auxiliary radiator. An on/off pump controlled by FCP inlet temperature will be needed to circulate the auxiliary radiator fluid, and a heat exchanger will be designed to transfer excess heat from the operating fuel cells. Since this loop will be inactive during normal fuel cell operation it is necessary to use a gas coolant (such as hydrogen) which will not freeze during inactive periods.

FUEL CELL/SPACECRAFT INTEGRATION

The major portion of the EPS radiators will occupy the same area now allotted for radiators in the Block II SM fairing. Additional area required for heat rejection due to the increased gross power capability of the system will be allotted on the SM surface. The size of the area and the optimum location will be determined in future studies. The Block II radiator locations 1 through 8 will be assigned to AES panels in the same order. The panels in the fairing are numbered 1 through 8 in a counter clockwise direction when looking aft. Panel No. 1 is above sector V and No. 8 is above sector IV. Numbers will be assigned to the additional panels when they are established.

The four powerplants will be located in the SM at station = 307.27. Three of the modules will be installed in sector IV and one module in sector I. The powerplants may be identified as power plant Number 1 through 4. Powerplants 1, 2, and 3 will be located in sector IV and powerplant 4 in sector I. The location of the three powerplants on the shelf in sector IV will be the same as Apollo Block II. The location of powerplant 4 will be determined after a detailed study of equipment location and access provision has been completed. The locations assigned to the powerplants will be used in establishing installation and checkout procedures and sequencing powerplant startups during a mission.

Each powerplant in the system weighs 248 pounds. Ten pounds of accessories are required to start each standby unit. For the four powerplant systems, 1005 pounds of powerplants and 20 pounds of accessories are required. The four powerplants, plus the start-up accessories for two powerplants, will occupy 37.6 cubic feet in the service module.

The Block II requirements for instrumentation and controls will be expanded for the four powerplant systems. The instrumentation and controls necessary to execute an in-space start will be established and incorporated in the requirements during future studies.

FUEL CELL TRADE-OFF SUMMARY

The trade-off summary analysis is separated into four categories: (1) configuration definition; (2) technology analysis; (3) reliability; (4) cost analysis. Results of all four categories are given in Table 42.

Since powerplant configurations IA, IB, IIA, IIB, IIIA, IIIB, and IIIC are only minor modifications of other configurations, or included hot standby units (which were dropped from further consideration very early in the study), they are not included in Table 42.

From results given in the technology analysis section (includes realibility) of Table 42, the three basic powerplant configurations IC, IIC, and IID are chosen for further study on basis of technology, reliability, and cost. In addition, they are considered to be within near-minimum change constraints which are imposed upon the study.

By comparing (1) mission power requirements, and (2) technology, reliability, and cost considerations of the various proposed powerplant configurations, the mission-reliability requirement of 0.995 was found to be met by (1) 400-hour modules (two operating with four cold-standby) and (2) 1200-hour modules (two operating with two cold-standby). Cost comparisons indicated that development cost of 1200-hour modules would be more than offset by lower delivery cost occasioned by the fewer number required. Also, when compared on the basis of 1600-watt average power, as given in Table 42, the 1200-hour module configuration weighs only 2820 pounds (including fuel supply), or 530 pounds less than the 400-hour configuration. Since the 1200-hour module configuration has substantially equivalent reliability, costs less, and weighs less, it is clearly more desirable for AES missions greater than 14 days.

Configuration IID produces 4000 watts, which is 1120 watts greater than configuration IIC, for an estimated subcontractor program cost of $36.001 million — $2.581 million more than the comparable cost for IIC. The problem of selection was thus reduced to comparing advantages accruing to greater power-producing capability against its greater cost in dollars. The problem was approached by analyzing power and energy relations of each configuration as it would be used with peaking and startup batteries in AES mission number 3, and a hypothetical worst-case mission using the battery sized for the former. The results of this analysis indicated that configuration IID would provide the greater flexibility to meet changing AES mission power demands and would require less weight penalty for peaking batteries and for battery recharging reactants. These considerations, plus the greater mission success and crew safety reliability of configuration IID, resulted in the selection of system IID for future study.

T.

APPROACH	I			
CONFIGURATION	C	D	C	
SYSTEM:				
OPERATING	2	2	2	
STANDBY	4	4	2	
STANDBY CONDITION	COLD	HOT	COLD	C
POWERPLANT:				
CHANGES TO PC3A-2, BLOCK II				
CERIA COATING	—	—	X	
CERIA FILL	—	—	—	
ACTIVATION	—	—	—	
INCREASED CELL SINTER AREA	—	—	—	
NUMBER OF CELLS	—	—	—	
VOLTAGE LIMITER	—	—	—	
TWO-STEP REGULATOR	X	—	X	
A.S.P.	X	X	X	
28 VOLT HEATER	X	X	X	
PR. REGEN. BYP. CONT.				
RESET CAPABILITY	—	—	—	
AVERAGE POWER	1600 2300	1600 2300	1600 2300	160S
SYSTEM WEIGHT AND VOLUME				
WEIGHT, LB				
TOTAL	3349 4188	3571 4442	2833 3672	292
REACTANTS & TANKS	1821 2660	2059 2930	1821 2660	179
POWERPLANT & ACCESSORIES	1352.4	1342.8	901.6	9
MOUNT STRUCTURE	137.4	137.4	91.6	
ACCESSORIES	38.0	32.0	19.0	
VOLUME, CU. FT				
TOTAL	120.4 152.6	130.2 163.8	101.8 133.6	10
POWERPLANT & ACCESSORIES	56.2 56.2	56.2 56.2	37.6 37.6	37
TANKS	64.2 96.4	74.0 107.6	64.2 96.0	64
SYSTEM PERFORMANCE AND RELIABILITY				
POWER, WATTS				
MAXIMUM	3180	2980	3180	
AVAILABLE DURING RECOVERY	1590	1490	1590	
TIME TO POWER RECOVERY, HOURS	2.8	.15	2.8	
TEMPERATURE, °F				
POWERPLANT OUTPUT				
800 WATTS	425 —	423 —	425 —	38
1150 WATTS	— 434	— 433	— 434	—
MAXIMUM POWER	445	442	445	
RELIABILITY	0.9981	0.9981	0.9994	0
CREW SAFETY RELIABILITY	>.9(6)55	>.9(6)55	0.9(5)87	0
EXPECTED POWERPLANT LIFE, HOURS	500	500	1200+	1
PROGRAM ANALYSIS				
COST, $10*				
TOTAL	38.408	36.411	33.419	3
DEVELOPMENT	3.125	2.192	6.721	
DELIVERY	29.091	27.614	21.420	2
OTHER	6.192	6.605	5.278	
NUMBER OF DELIVERIES				
FCP	138	138	92	
ASP	92	92	46	
DEVELOPMENT TEST HOURS				
POWERPLANT	1000	400	2945	
STACKS	—	—	7000	
CELLS	—	—	16250	
OTHER	12175	3600	12175	
ASP	4875	3760	4875	
TIME TO START OF QUALIFICATION TEST, MONTHS	19	19	19	

Table 42. Decision Information Table

	I		II				III		IV		
	C	D	C	D	E	F	D	E	A	B	C
	2	2	2	2	2	3	2	2	2	2	2
	4	4	2	2	1	0	1	0	1	0	0
		HOT	COLD	COLD	COLD	—	COLD	—	COLD	—	—
	—	—	X	X	X	X	—	X	—	—	—
	—	—	—	—	—	—	—	—	X	X	—
	—	—	—	X	X	X	X	X	X	X	X
	—	—	—	—	—	—	X	X	—	X	X
	—	—	—	—	—	29	29	28	—	28	26
	—	—	—	X	X	X	—	X	X	X	—
		X	X	X	X	—	X	—	X	—	—
		X	X	X	X	—	X	—	X	—	—
	—	—	—	—	—	X	—	—	—	—	—
	2300	1600 2300	1600 2300	1600 2300	1600 2300	1600 2300	1600 2300	1600 2300	1600 2300	1600 2300	1600 2300
		3571 4442	2833 3672	2820 3625	2559 3364	2756 3272	2552 3282	2402 2947	2469 3274	2356 2901	1981 2666
		2059 2930	1821 2660	1794 2599	1794 2599	2028 2544	1732 2462	1860 2405	1794 2599	1860 2405	1611 2296
		1342.8	901.6	913.6	685.2	657.3	741.3	494.2	595.2	448.2	334.0
		137.4	91.6	91.6	68.7	68.7	68.7	45.8	68.7	45.8	36.0
		32.0	19.0	21.0	11.5	2.0	9.5	2.0	11.5	2.0	—
		130.2 163.8	101.8 133.5	101.6 133.6	92.3 124.3	102.0 121.0	92.1 118.7	86.3 105.7	92.3 124.3	86.3 105.7	69.0 94.5
		56.2 56.2	37.6 37.6	37.6 37.6	28.3 28.3	28.0 28.0	28.3 28.3	18.7 18.7	28.3 28.3	18.7 18.7	10.0 10.0
		74.0 107.6	64.2 96.0	64.0 96.0	64.0 96.0	74.0 93.0	63.8 90.4	67.6 87.0	64.0 96.0	67.6 87.0	59.0 84.5
		2980	3180	4000	4000	4710	4000	5680	4000	5680	8000
		1490	1590	2000	2000	4000	2000	2840	2000	2840	4000
		.15	2.8	2.8	2.8	—	2.65	—	2.8	—	—
		423 —	425 —	382 —	382 —	400 —	375 —	421 —	382 —	421 —	420 —
		— 433	— 434	— 396	— 396	— 413	— 390	— 421	— 396	— 421	— 426
		442	445	426	426	429	408	448	426	448	460
		0.9981	0.9994	0.9998	0.9948	0.9907	0.9982	0.99916	0.9980	0.99916	0.99927
		>.9(6)55	0.9(5)87	0.9(6)71	0.9(5)02	0.9(4)81	0.9(5)81	0.9(4)23	0.9(5)77	0.9(4)23	0.9(4)33
		500	1200+	1200+	1200+	1200+	1200+	1200+	1200+	1200+	1200+
		36.411	33.419	36.001	30.703	27.898	36.826	28.587			
		2.192	6.721	8.338	8.338	7.242	13.459	11.615			
		27.614	21.420	22.213	17.300	16.047	18.235	12.478			
		6.605	5.278	5.450	5.065	4.609	5.132	4.494			
		138	92	92	69	69	69	46			
		92	46	46	23	—	23	—			
		400	2945	3445	3445	2895	11935	10735			
		—	7000	8750	8750	8750	21950	21950			
		—	16250	22900	22900	22900	32100	32100			
		3600	12175	24400	24400	18500	17100	11700			
		3760	4875	4875	4875	—	4875	—			
		19	19	13	19	19	19	19			

CONCLUSIONS

The study approach was to define the requirements for power, to conduct spacecraft configuration analyses and trade-off studies, and then, after certain spacecraft design decisions were made, to conduct trade-offs within the subsystem design study. A primary tool that was developed and used was a series of IBM 7094 computer programs that tabulate power load profile information and, by simulating various system designs and conditions, determine the system capability to perform a particular mission.

Reliability in a broad sense was also used as a design tool during this study phase. The use was not only by comparing calculated reliability for a system with the reliability goal or apportionment, but the severity and frequency of use (or stress) was determined for various components and compared with the predicted capability of that system to withstand that stress. Where potential problems were identified, alternative designs were analyzed to accomplish the function, as well as to investigate component improvement.

POWER REQUIREMENTS

Power requirements were tabulated for two extremes of power system operation, (1) housekeeping loads, or those required for the spacecraft to sustain itself in orbit but not to do experiments, and (2) the power requirements for the total mission, assuming the prime power source for the spacecraft/laboratory complex is in the Apollo CSM. The power loads for the first case are based on Apollo Block II subsystem design and are relatively accurately known. This load average is 681 watts ac and 473 watts dc provided two fuel cells (with a parasite load of 77 watts per fuel cell - the proposed Block II improvement) are operating and the average inverter efficiency is 75 percent. The power loads for the second case are based on the most severe of the four reference missions studied, which was the lunar mapping mission. This mission (reference mission 3) requires sustained peaks of 4011 watts as well as extended low power periods where the total load is sometimes down to 1380 watts.

ENTRY AND PYROTECHNIC BATTERIES

The requirements that the AES missions impose on the entry, postlanding, and pyrotechnic batteries were determined and compared with battery capability. For the entry batteries, it was found that, although the inherent shelf life (or charged storage time prior to discharge) is adequate for a 45-day mission, there are certain spacecraft loads continually connected to these

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batteries, and the energy consumed by these loads exceeds the allowable amount unless the batteries are recharged in flight. (The mission loads plus the wet-stand charge loss, plus the entry and landing loads exceed battery capacity.) Thus, for AES, crew safety depends on success in recharging these batteries during flight. Since the battery charger is not a crew safety item for Block II, its design reliability is not as high as is desired for AES. Redesign of this item or redundancy in this function should be investigated during the next AES phase.

From both a capacity and charge/discharge cycle standpoint, the entry batteries are inadequate to supply the peak load requirements. Therefore, another means of supplying these peaks is required. If the only peak loads imposed on the entry batteries are those associated with firing the SPS engine (as is Block II), the entry batteries can be used; however, if supplementary batteries are onboard the spacecraft for other purposes, it is recommended that they be used for SPS firing also.

It was determined that the charged stand-time capability of the pyrotechnic battery is inadequate for AES missions, and that these batteries be redesigned to incorporate separators capable of a 60-day or longer charged standtime.

POWER FOR PEAK LOADS

The peak loads of reference mission 3 could possibly be supplied by operating three, 2000-watt fuel cells or by supplementing the power of two fuel cells with battery power. Operating three 2000-watt fuel cells in parallel has the disadvantage of the additional parasite load of the fuel cell pumps and the load dissipated in the voltage limiter, in addition to the reliability degradation caused by fuel cell operation rather than standby.

The system simulation studies show that approximately 300 pounds of reactants are saved by using a 186-pound supplementary battery system for reference mission 3. This same battery system can also be used to start a cold fuel cell in case an operating fuel cell suddenly and completely fails, so that fuel cell start-up energy is not available from the CSM bus.

The supplementary battery system recommended consists of two 70-ampere-hour silver cadmium secondary batteries and two battery chargers. They are located in the service module.

POWER CONVERSION

Examination of AES ac loads indicate that the Block II inverter has ample capacity to meet AES requirements. However, the inverter reliability, particularly for crew safety, is below the desired level. Considerable study was made on the cost-effectiveness of reliability improvement, both by NAA and,

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on a voluntary basis, by Westinghouse. These studies established that, if overall reliability improvement is required, it is more desirable to improve the inverter itself than to add a fourth inverter.

FUEL CELL TRADE-OFF STUDIES

The basic fuel cell trade-off was between use of the 400-hour Apollo Block II powerplants and improving the fuel cells to the extent necessary to extend normal fuel cell wearout beyond the AES mission durations. Many separate modifications to the basic PC3A-2 powerplant were considered to provide 1200-hour life and other desirable characteristics; these modifications were grouped into 18 powerplant configurations which were compared with each other and with systems using the 400-hour fuel cell. The studies showed that significant weight and system operation advantages were obtained by using 1200-hour powerplants. The study also showed that a cost advantage was obtained by using certain 1200-hour powerplant configurations, since the development cost was offset by decreased unit costs per spacecraft. Therefore, it was recommended that a 1200-hour fuel cell be developed.

The most effective means available to obtain 1200-hour life in a P&W fuel cell with minimum change to the FCP configuration are by lowering operating temperature, ceria coating of the hydrogen electrode, and improved manufacturing processes to ensure greater sinter integrity. Increased performance characteristics (again with minimum change) can be obtained by increased operating temperature, cobalt activation of the oxygen electrode and palladium activation of the hydrogen electrode, increased cell area, and changes in the primary regenerator bypass valve. In keeping with overall AES program ground rules of minimum change to Apollo systems, the least amount of development that would provide 1200-hour life and retain Apollo FCP performance characteristics, ceria coating, was recommended for the AES configuration. In addition, cell activation, a change not absolutely necessary for 1200-hour life but nevertheless providing performance improvement that could be used to lower normal operating temperature and/or increase capacity, was recommended. Thus the recommended P&W fuel cell powerplant is the one requiring minimum change from the PC3A-2 but providing 1200-hour life and 2000-watt power capability.

The performance characteristics of the AES fuel cell are such that reduction in the number of cells or else a voltage limiter is necessary to keep the voltage within spacecraft limitations at minimum load. It is recommended that a voltage limiter be provided for this purpose. This recommendation is based primarily on a desire to minimize changes to the FCP itself and to maintain the 2000-watt capability. With two fuel cells in operation, the voltage limiter is seldom used because AES normal continuous loads are high enough to prevent the over-voltage condition. It would, however, prevent damage to spacecraft components in case of a drop in the power demand.

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FUEL CELL STARTUP

In a power system design where the start of redundant units is expected in a normal mission, it would be very desirable to have a unit that could essentially be turned off and on. With the P & W fuel cell, this type of operation could be approached by an automatic startup and shutdown programmer (ASSP) coupled with a catalytic reactor for FCP heating. In this way, no significant sharing of spacecraft power or alternative use of peak power batteries would be required. However, for AES, the low probability of having to start a FCP, and the even lower probability of needing to start one quickly after another has completely failed, makes the development of a catalytic reactor seem undesirable, particularly when considering the high cost and high development risk of such a device. It is therefore recommended that FCP startup heaters be modified for use with 28-volt power, and that startup be accomplished with spacecraft power and/or the supplementary batteries.

EPS RADIATORS

The radiators for Apollo Block II are designed for operation with three fuel cells that are operating from the time of launch. Each fuel cell has its own independent radiator loop, and there is no attempt to keep the loop from freezing after a FCP has been shut down.

The AES fuel cell design requires a different radiator concept because, (1) there are four rather than three fuel cells, (2) only two normally operate at any one time, (3) the redundant powerplants are started in space if necessary, and (4) the fuel cells have 2000-watt capability rather than the Block II capability of 1420 watts. Several alternative radiator schemes were analyzed, including common heat rejection loops, dual radiator fluids, and auxiliary radiators. It was determined that the problem of freezing inactive loops can be avoided by close spacing of the four tubes in each panel; the overall heat rejection effectiveness of the method is reduced only 10 percent from the Block II design of equally spaced tubes.

The radiator area on the service module fairing that is used by Block II is about the maximum that can be used, since the rest of the fairing area must be removable for equipment access. With two powerplants operating, the use of only this area limits FCP output to about 2900 watts in lunar orbit. In order to provide cooling for 4000-watt operation, an auxiliary or supplementary radiator was recommended. This radiator system would pick up heat from fuel cell glycol loops via a small heat exchanger in the radiator return lines. When the temperature of the fluid exceeds a set limit, the auxiliary radiator pump will start so that excess heat is rejected by the auxiliary radiator. In this concept, the auxiliary radiator and its components are not vital to crew safety or even mission success with the exception of mission

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power loads beyond 2900 watts in lunar orbit. The radiator can be easily removed without fear of contaminating fuel cells, and only two connections to the auxiliary radiator panel are required, regardless of the number of fuel cells. The auxiliary radiator contains a fluid which freezes at only very low temperatures and is fabricated into the panel in the same way as Apollo Block II radiator panels. It is recommended that this radiator be located on Sector IV only, with the spacecraft operation restriction that Sector IV not be held at worst-case orientation (toward the lunar surface) at the same time that extended power peaks (beyond 2900 watts) are required by the mission.

SYSTEM CAPABILITY

The overall power system design objective is to provide the maximum power capability, energy capability, and power system flexibility that is possible, consistent with minimum change ground rules. As is shown in Figure 41, the proposed system allows the total spacecraft load profile to vary at random from approximately 1600 watts to 4000 watts with efficient utilization of reactants. For short periods of time (with some penalty in KWH efficiency) the load can vary from zero to the maximum that the distribution system can carry, which is approximately 5630 watts. Peaks even beyond this level can be tolerated for a few minutes, depending on equipment location. This power load flexibility of about 4500 watts (from the self-sustaining minimum of two fuel cells operating to the maximum 1-hour continuous power) corresponds to Block II power load design flexibility of only 1150 watts. Missions can be planned that utilize the total energy contained in the cryogenic storage system via any load profile within these broad limits and within the mission duration limitations imposed by the cryogenic storage system insulation effectiveness.

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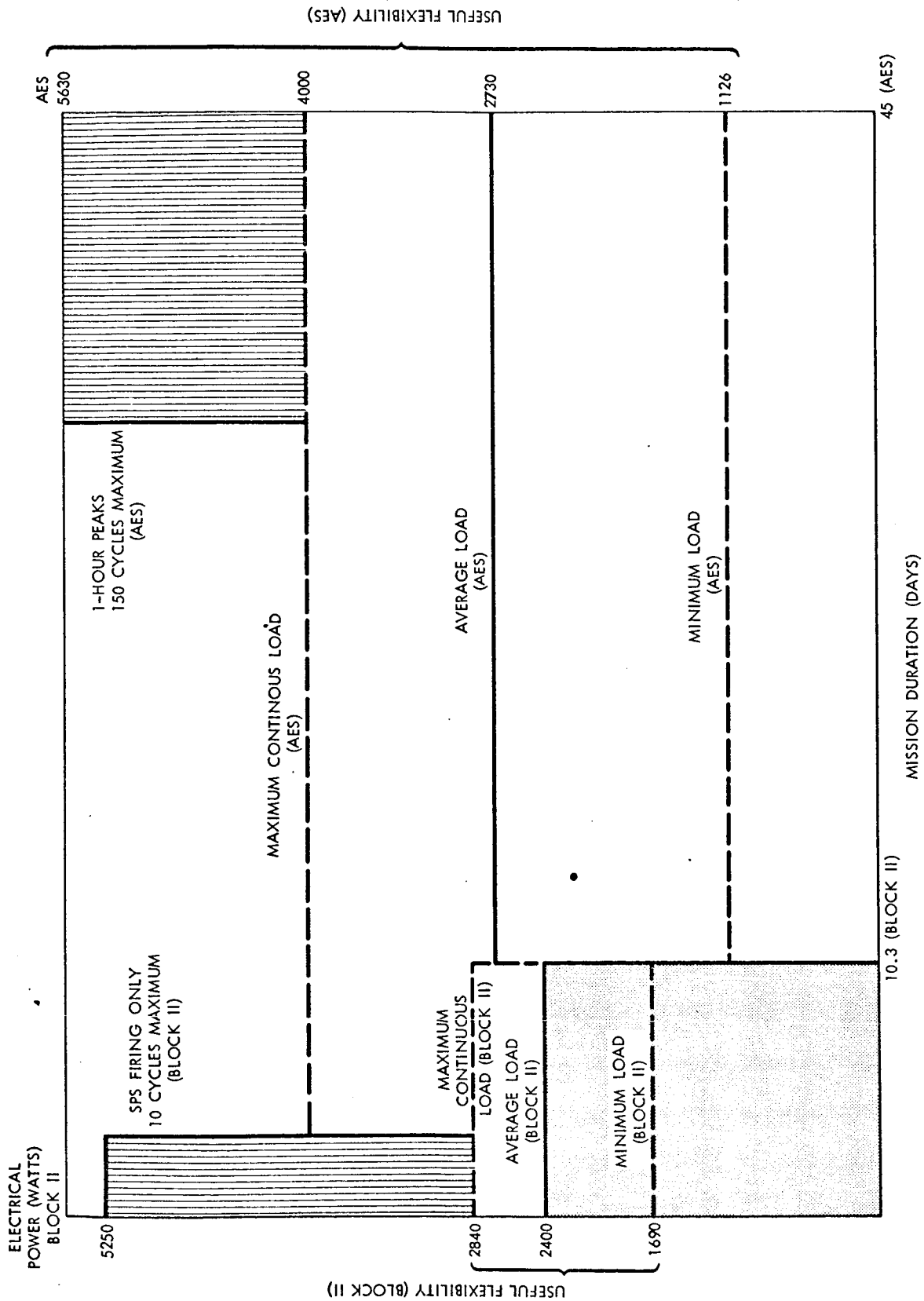
Figure 41. Electrical Power Comparison

CRYOGENIC STORAGE SYSTEM
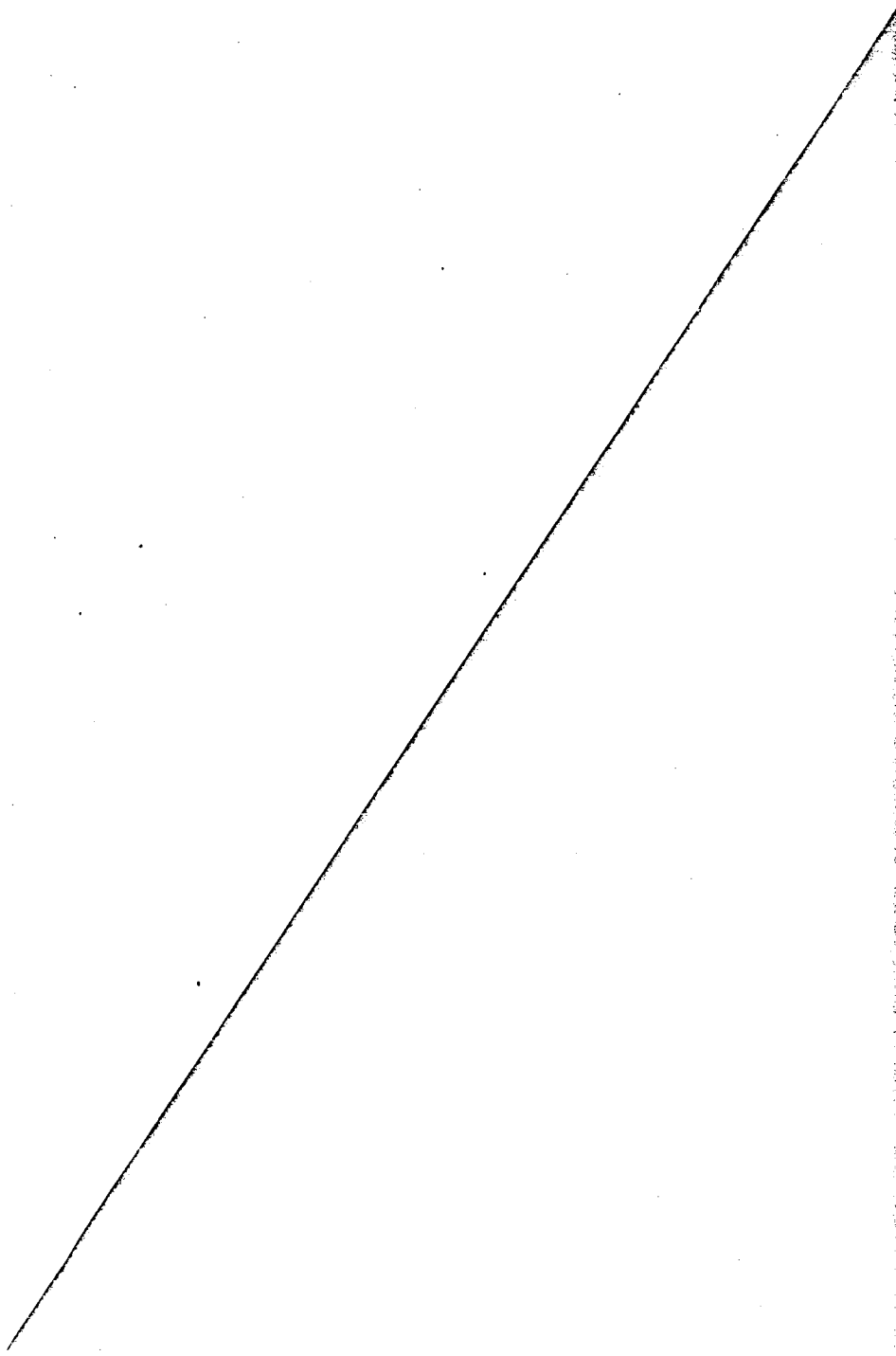
CRYOGENIC STORAGE SYSTEM

The AES cryogenic gas storage system (CGSS) supplies hydrogen and oxygen to the fuel cell power system and also oxygen and nitrogen for the environmental control system (ECS). It is thus the prime source of energy for spacecraft orbital operations. The major effort in the CGSS area was concentrated toward the definition of a new system that would provide operation for up to 45 days. Studies were based on results of the previous AES study phase and on the guidelines specified by NASA. Initially, spacecraft configuration tradeoff analyses were made on two tank sizes, one storing quantities for CSM housekeeping functions only and the other storing the maximum quantity that could be reasonably fitted into sectors I and IV of the SM. These tradeoff analyses were based on requirements as they were known at the beginning of the study and resulted in the recommendation of a particular spacecraft configuration and two CGSS tank sizes — both within the general category of maximum volume. These two approaches were then analyzed in detail to determine dimensions, installation, weight, performance, and other design characteristics. As requirements changed, these were reflected in the system definition activity. The design goal of the baseline study was to maximize the energy that could be supplied to the entire orbital spacecraft and thus provide maximum payload capability (ability to power experiments) for each flight.

In addition to analyses of particular CGSS designs, several promising alternatives were explored, including subcritical storage, alternative materials and process investigations, and a "half quantity" tankage arrangement whereby a lunar orbit mission could be performed with SM sector I empty for experiments.

The following section presents a summary of the CGSS studies performed during the AES preliminary definition phase. For a more detailed description of the cryogenic storage system studies, see SID 65-1526.

AES REQUIREMENTS

During the first half of the study, one set of requirements was used to develop subsystem trade-off data. However, as new data became available, some of these requirements were changed, and a new set of requirements was introduced.

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INITIAL REQUIREMENTS

The initial requirements were for two tank configurations. One set of tank configurations was sized for a nominal 45-day mission based on CM electrical power system and environmental control system housekeeping requirements, and a second set using the maximum volume available in sectors I and IV, was sized. In the maximum-volume case, the cryogenic storage system supplied all of the oxygen requirements for leakage and metabolic consumption in the CSM and the external module and all oxygen and hydrogen reactants for all electrical power required during the mission.

The fluid requirements were determined on the basis of a 7.0 psia, 50-percent oxygen and 50-percent nitrogen atmosphere. The average electrical power requirements used for the housekeeping case were 1600 watts. The design goal for the maximum volume case was 2840 watts, since that power level represented the maximum power output of two Apollo Block II fuel cells and represented approximately twice the net housekeeping power load. The calculated fluid quantities for each system configuration for the initial requirements are shown in Table 43.

FINAL REQUIREMENTS

As a result of the subsystem trade-off at the midpoint of the study, the following set of requirements and ground rules was established:

1. For 45-day missions, the crew compartment atmosphere should be 5 psia, 70-percent oxygen and 30-percent nitrogen. The capability to operate alternatively at 5 psia, pure oxygen must be designed into the system. Diluent tankage will be added in SM sectors II and VI. Laboratory metabolic and leakage requirements will be supplied from the CSM.

2. The baseline fuel-cell system should be Pratt and Whitney, 1200-hour cells with in-flight start and cell activation. Three fuel cells will be located on the existing shelf in sector IV, and a fourth cell will be located on a similar shelf added to sector I.

Table 43. Cryogenic Storage Fluid Requirements

System Configuration	Type Fluid (lb)		
	Oxygen	Hydrogen	Nitrogen
Housekeeping Function Size	1425	160	165
Maximum Volume Size	3000	315	300

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3. The CSM should not require power from the laboratory module. However, provisions will be made to provide both ac and dc power to busses in the laboratory and SM sector I.

4. Supercritical cryogenic storage of oxygen and hydrogen should consist of one tank for each in sector IV and an identical set of tanks in sector I. The baseline tank configuration should incorporate hemispherical domes and be sized for maximum volume without penetration of the SM aft bulkhead. Trade-off data will be developed on supercritical versus subcritical cryogenic storage as well as on dome shape (that is, hemispherical versus elliptical) and aft bulkhead penetration.

5. The sector I fuel cell and tank installation should be easily removable up to the point of cryogenic loading at the launch pad to permit the alternative installation of the pallet or other experimental devices. This would result in reduced mission durations, when only the power provided by the fuel cells and cryogenics remaining in sector IV would be used. In this case, an emergency gaseous oxygen and hydrogen supply should be installed for crew safety. These will be cylindrical tanks installed below the RCS propellant tankage in SM sectors II, III, and V.

An evaluation was made of the Block II environmental requirements; and two changes, vibration level and environmental temperature, were identified. The Block II vibration levels are being revised upwards and will require at least two synchronized shakers to produce the required force for AES oxygen-tank qualification. The environmental temperature in sectors I and IV were calculated for earth orbital and lunar orbital missions for different orientations and passive thermal control modes. The maximum oxygen tank outer shell temperature was 80 F, and the maximum hydrogen tank outer shell temperature was 150 F. These temperatures are preliminary, and the Block II upper temperature limit of 170 F was therefore included in all heat-leak analyses that were performed for a range of temperatures from 80 F to 170 F.

The system flow requirements were determined and are shown in Table 44. The minimum flow rates are based on metabolic consumption of three men and no leakage. A 1200-watt housekeeping power level was used. As shown in Figures 42 and 43, the AES fuel cell reactant consumption remains constant below 800 watts because of the voltage limiter. Based on the inherent characteristics of the presently defined fuel cell and the present plan to operate two fuel cells at the same time, the system minimum flow rates could correspond to 1600 watts power level. The insulation analysis was performed for a range of minimum flow rates corresponding to the 1200-watt power level up to twice that flow rate.

Table 44. Cryogenic Storage System Flow Requirements (lb/hr)

Function	O₂ Max (1)	O₂ Min (1)	H₂ Max	H₂ Min	N₂ Max	N₂ Min
Environmental Control System						
Metabolic Consumption (2)	0.25	0.25	—	—	—	—
Leakage CM Only	0.20 (0.153)*	0	—	—	(0.53)*	0 0
Leakage CM and Lab	0.50 (0.371)*	— 0	—	—	(0.139)*	— 0
EVA & Emergency Repressurization	7.50	—	—	—	6.0	—
Total ECS Flow	8.25	0.25	—	—	6.139	No Flow
Electrical Power System						
Consumption 4000 watts	3.02	0.760	0.376	0.096	—	—
1200 watts	—	—	—	—	—	—
Purge (3)	0.60	0.005	0.70	0.007	—	—
Total EPS & ECS Flow	11.87	1.015	1.076	.103	6.139	No Flow
Recommended Design Flow (4)	12.0	1.0	1.0	.10	6.2	No Flow

NOTES: (1) 5 psia pure O₂ atmosphere *(2 gas atmosphere; 5 psia, 70% O₂, 30% N₂).

        (2) Single cryogenic O₂ tank in Sector IV is assumed for LEM escort mission which has lower rate.

        (3) Fuel cell purge average flow based on 0.6 lb/hr for O₂ and 0.75 lb/hr for H₂ for 2 fuel cells for 2 minutes each every 7 hours.

        (4) It is recommended that the minimum flow rates be revised to correspond to 1600 watts minimum power when temperature environment is fully defined.

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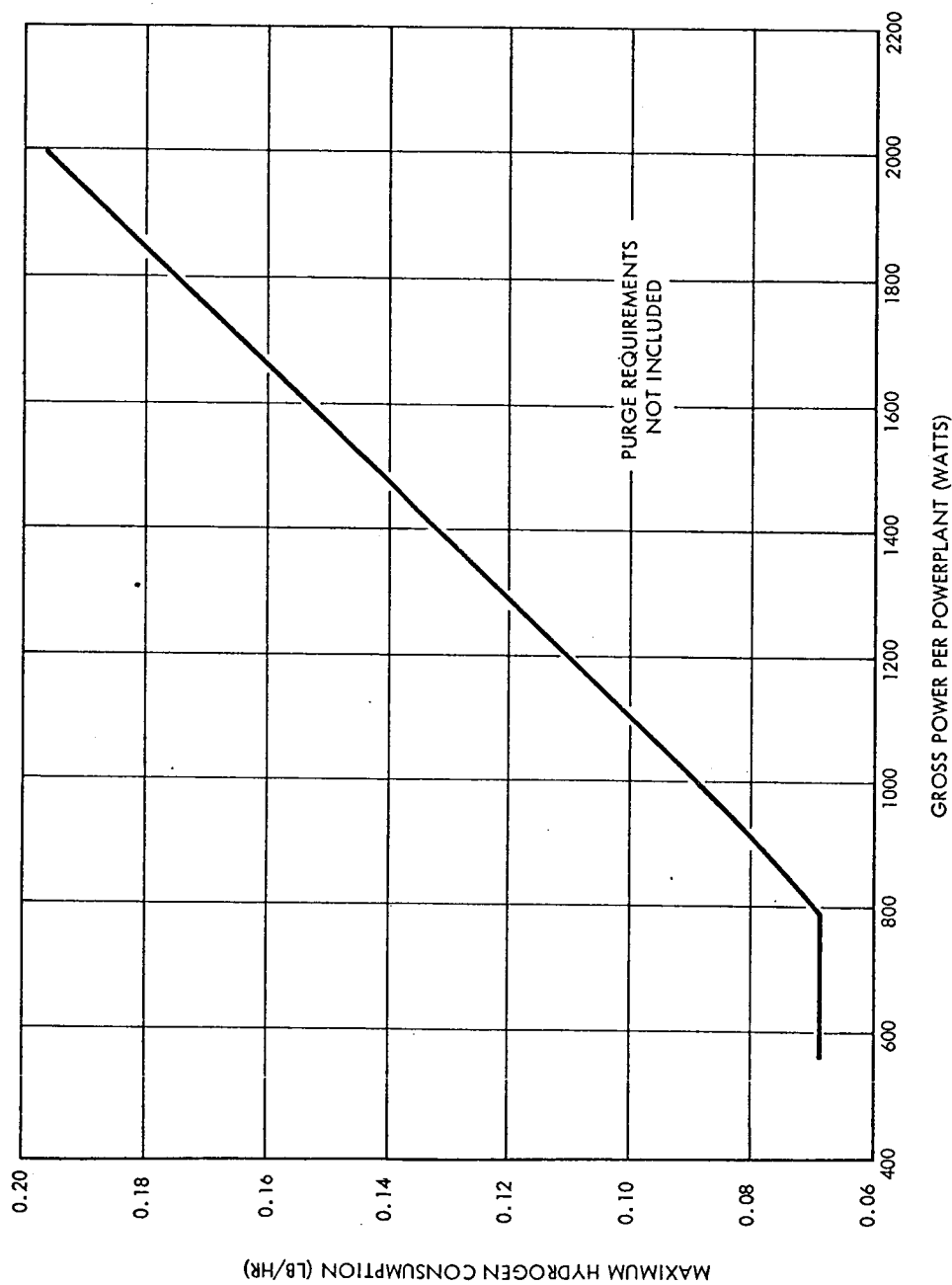
Figure 42. Steady-State Hydrogen Consumption

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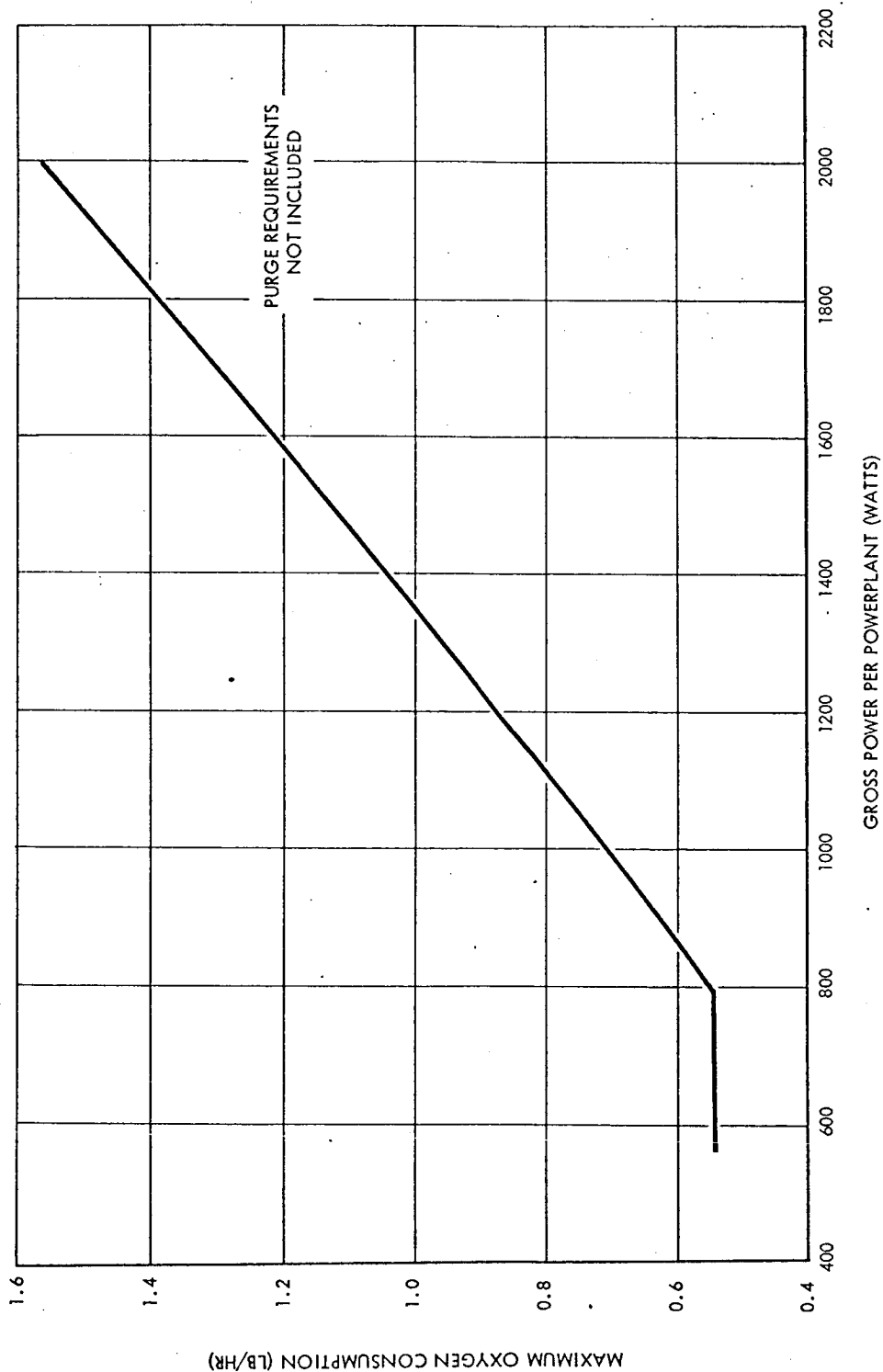
Figure 43. Steady-State Oxygen Consumption

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The fluid requirements for ECS were determined and are shown in Table 45. The electrical power conversion fluid quantities for the reference missions were determined but were not used in tank sizing. Rather, the objective of the study was to determine the maximum amount of reactants that can be stored in sectors I and IV of the service module.

Table 45. ECS Oxygen and Diluent Requirements

Oxygen Requirements for a 45 Day Mission – 5 psia, 70% Oxygen, 30% Nitrogen or Helium Atmosphere

Function	Diluent Type	
	Nitrogen	Helium
Metabolic Oxygen	270 lb	270 lb
Leakage, CM Plus Lab Plus Interlock	403 lb	493 lb
EVA, 7 Repressurizations	42 lb	42 lb
3 Emergency Repressurizations, CM Plus Lab	32 lb	32 lb
Total Oxygen Required	747 lb	837 lb

Diluent Requirements for a 45 Day Mission – 5 pisa, 70% Oxygen, 30% Nitrogen or Helium Atmosphere

Function	Diluent Type	
	Nitrogen	Helium
Leakage, CM Plus Lab	118 lb	21 lb
EVA, 7 Repressurizations of CM	12.4 lb	1.8 lb
3 Emergency Repressurizations, CM Plus Lab	9.3 lb	1.3 lb
Total Diluent Required	139.7 lb	24.1 lb

Oxygen and Hydrogen Requirements for Abort From Lunar Orbit

Function	Oxygen	Hydrogen
ECS		
Metabolic Consumption, 3 Men (2 lb/day)	27.5 lb	—
Leakage, CM Only (.2 lb/hr)	22.0 lb	—
EPS		
Consumption @ 1200 watts, one fuel cell	92.5 lb	11.5 lb
Purge	.5 lb	.7 lb
Total Fluid Required for Abort From Lunar Orbit	142.5 lb	12.2 lb

TANK SIZING AND WEIGHT

As previously mentioned, two sets of ground rules for tank sizing were used; one set was used for development of subsystem trade-off data, and a second set was introduced at the midpoint of the study for further development of trade-off data and preliminary definition. Each set of ground rules and resulting tank sizing data is treated separately.

INITIAL TANK SIZING

The housekeeping size tanks as shown in Figure 44 fit easily into the service module and present no problem as far as envelope dimensions are concerned. The analyses of the maximum volume size tanks, however, show that the tank overall length for a given volume has a drastic effect on system weight as shown in Figure 45. A weight saving of 482 pounds could be realized by extending the tankage through and below the aft bulkhead, as shown in Figure 46, rather than storing the same quantity completely within the sector. Extending the tankage through the aft bulkhead would also permit all outer shell domes to be identical for all tanks and the pressure vessel domes to have the same dome height (1.4:1 ellipse) which would minimize tooling requirements. A further 15-percent increase over the baseline maximum volume size is possible with both maximum penetration of the aft bulkhead and elliptical domes. The use of 1.4:1 elliptical domes without bulkhead penetration results in eight-percent smaller volume than the baseline maximum volume case. These characteristics and other results of the initial tank sizing trade-off study are shown in Table 46.

The following two types of N_2 storage were considered: (1) cryogenic storage in one or two tanks and (2) high-pressure N_2 in two or more spherical or cylindrical tanks. A high-pressure storage, for the maximum size system, will be up to 365 pounds heavier than a cryogenic system but has higher reliability and a significantly lower cost, as shown in Table 47.

FINAL TANK SIZING

With the introduction at the midpoint of the study, of a 5 psia, 70-percent oxygen and 30-percent nitrogen cabin atmosphere and of a new baseline fuel cell a new tank size and weight trade-off analysis was performed to establish two tank sizes for preliminary definition. The ground rules specified that one configuration should include hemispherical domes and be sized for maximum volume without penetration of the SM aft bulkhead. Trade-off data was developed for tanks with bulkhead penetration and tanks

NOTE:

FULL-UP SPS WITH HOUSEKEEPING H_2, O_2, N_2

RCS: ANY CHOICE WORKS WITH THIS CONFIGURATION

NO SPS ENGINE ACCESS
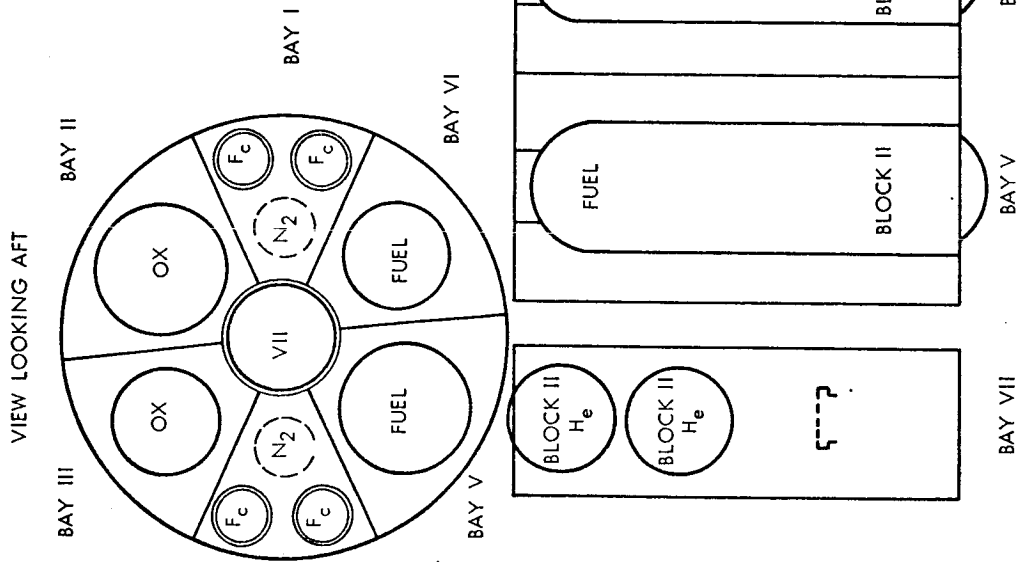
Figure 44. Cryogenic Storage System Sizing for Housekeeping Loads

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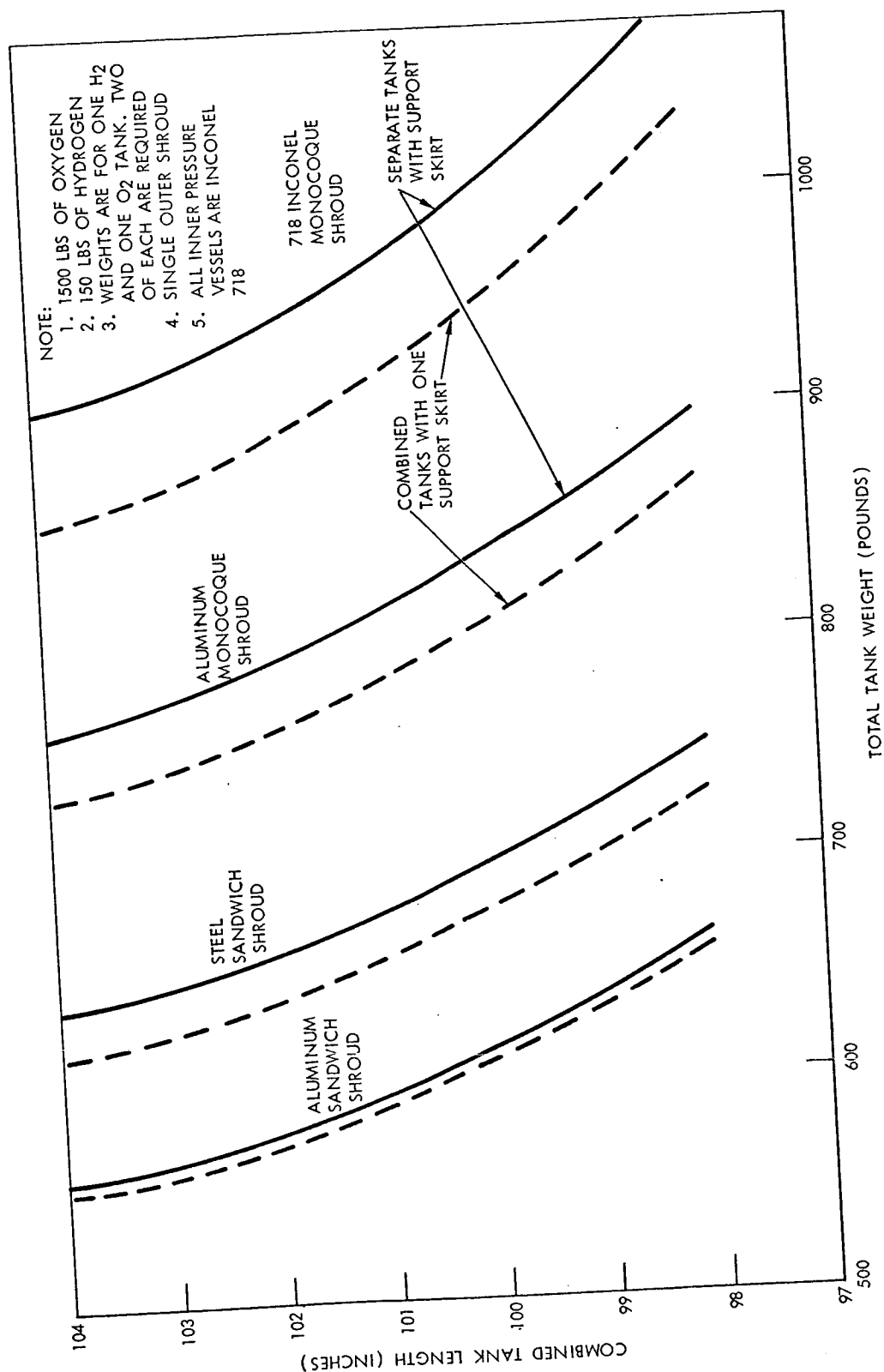
Figure 45. Comparative Weight of Hydrogen and Oxygen Tanks for Various Construction Methods

NOTE:
1. 1500 LBS OF OXYGEN
2. 150 LBS OF HYDROGEN
3. WEIGHTS ARE FOR ONE H2 AND ONE O2 TANK. TWO OF EACH ARE REQUIRED
4. SINGLE OUTER SHROUD
5. ALL INNER PRESSURE VESSELS ARE INCONEL 718

718 INCONEL MONOCOQUE SHROUD

SEPARATE TANKS WITH SUPPORT SKIRT

COMBINED TANKS WITH ONE SUPPORT SKIRT

ALUMINUM MONOCOQUE SHROUD

STEEL SANDWICH SHROUD

ALUMINUM SANDWICH SHROUD

TOTAL TANK WEIGHT (POUNDS)

COMBINED TANK LENGTH (INCHES)

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FULL-UP SPS WITH MAXIMUM VOLUME CRYOGENIC STORAGE

RCS: ANY CHOICE WORKS WITH THIS CONFIGURATION

NO SPS ENGINE ACCESS

N_2 GAS EITHER GAS STORAGE OR CRYO STORAGE (CRYO WILL USE SPHERES IN PLACE OF CYLINDER OF SAME DIAMETER

THE BASELINE MAXIMUM VOLUME CRYOGENIC STORAGE IS SHOWN WITH PENETRATION OF AFT BULKHEAD FOR MINIMUM WEIGHT
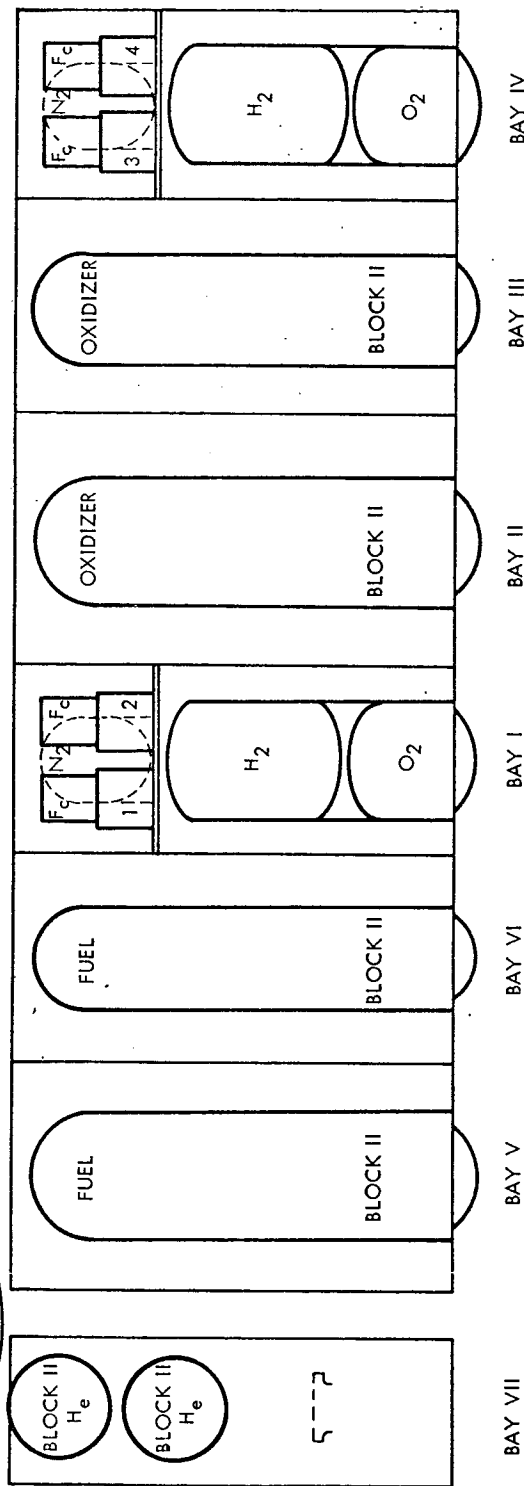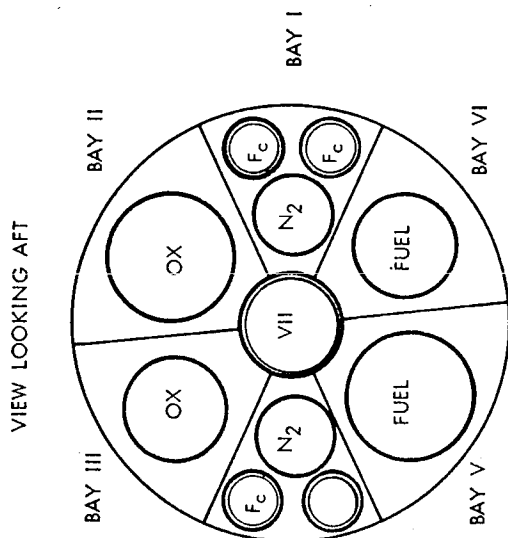
Figure 46. CGSS Maximum Volume with Bulkhead Penetration

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Table 46.　45-Day Cryogenic Oxygen and Hydrogen Systems Summary

	Housekeeping Size		Maximum Volume Sizes					
			Oxygen			Hydrogen		
	Oxygen	Hydrogen	Baseline Maximum Volume	Baseline +15%	Baseline -8%	Baseline Maximum Volume	Baseline +15%	Baseline -8%
Usable Fluid Weight (pounds)	1425	160	3000	3450	2780	315	345	290
Dome Configuration[1]	Sphere	Sphere	1.4:1 Ellipse	1.4:1 Ellipse	1.4:1 Ellipse	1.4:1 Ellipse	2.6:1 Ellipse	1.4:1 Ellipse
Material								
Inner Pressure Vessel	Inconel 718	Inconel 718	Inconel 718	Inconel 718	Inconel 718	Inconel 718	Inconel 718	Inconel 718
Outer Shell	Inconel 718	Inconel 718	Inconel 718	Inconel 718	Inconel 718	Inconel 718	Inconel 718	Inconel 718
Outside Diameter (inches)	36.2	41.5	41.5	41.5	41.5	41.5	41.5	41.5
Outside Length (inches)	36.2	45.25	40.0	44.25	39.0	64.5	62.75	59.0
Tank Dry Weight (pounds)	420	390	792	992	675	1008	1262	920
Tank Wet Weight (pounds)	1916	558	3942	4615	3594	1339	1625	1225
Maximum Power Level for 45 days (watts)	1600	1600	2840	3250	2610	2840	3250	2610
Estimated Cost[2] per Ship Set	$300,000	$296,000	$346,000	$350,000	$344,000	$321,000	$325,000	$320,000

Notes:　1　The baseline maximum volume size tanks shown penetrate aft bulkhead for minimum weight.
　　　　　2　Cost data is for comparison only.

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Table 47.　45-Day Nitrogen Storage Systems

Data Unit	Configurations			
	Housekeeping		Maximum Volume	
	High Pressure	Cryogenic	High Pressure	Cryogenic
Usable Fluid Weight	165 lb	165 lb	300 lb	300 lb
Tank Dry Weight	275 lb	140 lb	585 lb	220 lb
Tank Material	Ti 6A1 4V	Inconel 718	Ti 6A1 4V	Inconel 718
Number of Tanks		2		
Compartment Temperature	170 F	250 F	250 F	
Mission Success Reliability	0.999336	0.99798	0.999666183	0.99798
Estimated Cost Per Ship Set*	$25,000	$200,000	$55,000	$230,000
*Cost data are for comparison only.				

with elliptical domes. It was decided to include one additional factor for trade-off, that of increasing the tank diameter without penetration of the aft bulkhead. The trade-off point between bulkhead penetration and elliptical domes was selected so that the elliptical dome case has the same volume as can be obtained with maximum penetration of the aft bulkhead by hemispherical domes.

The tank size referred to as the baseline size is the maximum volume that can be obtained with hemispherical domes and without penetration of the SM aft bulkhead. It was determined that the present design of the radial beams limits the outside diameter of the tanks to 41.5 inches. The maximum combined length of one oxygen tank and one hydrogen tank should not exceed 100 inches if adequate clearance for installation of the tanks is to be provided. Preliminary definition studies of the storage tank established that the minimum practical dimension from the outside surface of the outer shell to the inside surface of the pressure vessel is 1 inch on the side, 1.6 inches at the bottom, and 2 inches at the top of the tank. These dimensions are required for the vapor-cooled shield, internal supports, and internal tubing, respectively. Minimum distance between the outside surfaces of the outer shell of

the oxygen and hydrogen tank is 0.3 inches. To provide adequate room for the internal support, the outer shell domes deviate substantially from the hemispherical contour. With the present design, the outer shell dome contour has a spherical radius of 26.7 inches with a 0.5 inch corner radius. The tank size, volume and weight obtained by each configuration is shown in Table 48. Cases 1 and 4 were selected for preliminary definition. The weights are based on Inconel 718 inner pressure vessels and aluminum monocoque outer shells. Block II limit pressure and structural safety factors were used.

Table 48. Tankage Trade-off Summary

Case	Pressure Vessel Dome Configuration	Tank Outside Diameter (inches)	Tank Length (inches) Oxygen	Tank Length (inches) Hydrogen	Tank Weight (pounds) Oxygen - (Two Tanks)	Tank Weight (pounds) Hydrogen - (Two Tanks)
Baseline configuration	Hemisphere	41.5	43.3	56.4	678	540
Increased diameter	Hemisphere	43.0	43.2	56.5	701	555
Baseline configuration	Hemisphere	41.5	45.9	61.8	739	607
Elliptical pressure vessel domes to obtain same volume as with bulkhead penetration	1.4:1 ellipse	41.5	41.9	57.8	917	675

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COMPARISON OF STORAGE METHODS

The following storage methods for hydrogen, oxygen, helium, and nitrogen were considered: (1) high-pressure storage, (2) supercritical cryogenic storage, and (3) subcritical cryogenic storage. All three methods do not apply to all of the fluids. For practical reasons, the trade-off was performed such that supercritical cryogenic storage is compared to subcritical cryogenic storage of oxygen and hydrogen except for the emergency (for lunar orbits with Sector 1 open) oxygen and hydrogen storage system which was explicitly limited to high-pressure storage. High-pressure storage of nitrogen and helium was compared to supercritical storage of these fluids.

SUPERCRITICAL VERSUS SUBCRITICAL STORAGE OF OXYGEN AND HYDROGEN

A reduction in operating pressure will usually result in lower system dry weight. It is, however, important to remember that the pressure vessel wall thickness can only be reduced to a certain point due to manufacturing considerations. At the present time, it appears that the inner pressure vessel wall thickness of the hydrogen tank is close to that limit. A pressure reduction in the hydrogen system would, therefore, not result in any appreciable weight reduction. The oxygen tank weight, however, can be reduced by a substantial amount in each case by lowering the limit pressure from the present Block II values.

For the purpose of this analysis, two new pressures, 850 psia and 600 psia, were selected. If the supercritical storage method is selected, 850 psia is the recommended system maximum operating pressure. If the subcritical storage method is selected, 600 psia is the recommended maximum operating pressure. This pressure was selected on the basis of 500 psia minimum operating pressure to assure determination of the proper abort quantity in lunar orbit by pressure and temperature measurement.

The oxygen system weight saving for the baseline configuration is approximately 50.4 pounds and 133.2 pounds for lowered pressure limits of 850 psia and 600 psia, respectively. An even greater savings is obtained for the elliptical configuration; 78.8 pounds and 196.8 pounds are saved for 850 and 600 psia pressure limits.

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CRYOGENIC VERSUS HIGH-PRESSURE STORAGE OF ECS DILUENT

A trade-off analysis was also made for the ECS diluent, considering both nitrogen and helium and high-pressure and cryogenic storage. The results of this trade-off are shown in Table 49 and were used to influence the overall ECS recommendation of nitrogen as the diluent gas. High-pressure storage of nitrogen is recommended, primarily because of the development cost of a cryogenic system.

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Table 49. ECS Diluent Storage Trade-off Summary

Location	HELIUM			NITROGEN		
	Cryogenic Sector I	High Pressure		Cryogenic Sector I	High Pressure	
		Sector I	SPS Sectors		Sector I	SPS Sectors
Environmental Temperature	250°F	250°F	100°F	250°F	250°F	100°F
Loading Temperature	-420°F	70°F	70°F	-320°F	70°F	70°F
Operating Pressure	2000 psia	3680 psia	3600 psia	900 psia	3680 psia	3600 psia
Material Inner	Inconel 718	Inconel 718	Ti 6Al-4V	Inconel 718	Inconel 718	Ti 6Al-4V
Outer	Aluminum	---	---	Aluminum	---	---
Size	25.5 in. O.D.	25.0x32.35	12.5x58.0	25.1 in. O.D.	27.3x43.0	12.5x46.5
Number of Tanks	1	2	3	1	1	3
Usable Fluid Wt.	24.1 lb.	24.1 lb.	24.1 lb.	139.7 lb.	139.7 lb.	139.7 lb.
Tank Dry Wt.	78 lb.	490 lb.	295 lb.	82 lb.	435 lb.	230 lb.
Tank Wet Wt.	103 lb.	515 lb.	320 lb.	225 lb.	575 lb.	370 lb.
Estimated Cost Shipset (Development not included)	$115.000	$20.000	$23.000	$115.000	$18.000	$22.000

Note: Cost data are for comparison only.

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STORAGE TANK DESIGN ANALYSES

The preliminary analyses, including material evaluation, stress, and thermal analyses, resulted in the following recommended design approach for the AES cryogenic oxygen and hydrogen storage tanks:

1. Forged, machined, and welded Inconel 718 inner-pressure vessels for both storage tanks

2. Moanocoque aluminum outer shells for both storage tanks

3. Vapor-cooled shield insulation with local supports (similar to the proposed Apollo Block II hydrogen storage tank).

4. Fan heaters for pressure control

5. Block II fill and vent valves

PRESSURE VESSEL AND OUTER SHELL

The recommended design approach is based on using Block II materials and fabrication processes, design concepts, and parts, where possible. The forged and machined pressure-vessel fabrication method is recommended on the basis of minimum development time and risk. Inconel 718 is recommended for both pressure vessels which will allow the use of one forging for all domes. Characteristics are shown in Table 50.

Aluminum monocoque outer shells are recommended primarily because aluminum offers the lowest weight and the lowest cost.

INSULATION AND INNER TANK SUPPORTS

Vapor-cooled shields and local supports will provide the required thermal performance at a low weight, good vacuum stability, and reproducible performance. The number of shields and plated surfaces will depend on final determination of environmental temperature and system minimum flow.

The general insulation method currently being used for cryogenic storage in space vehicles is multilayer Mylar or aluminum foil-type materials in an annular vacuum space. In addition, a vapor-cooled shield

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Table 50. Summary of Vessel Properties

Configuration A (Hemispherical domes)	Type of Vessel	Section		Material	Wall Thickness (in.)
	H_2	Tank	Cylinder	Inconel 718	0.055
			Head	"	0.028
		Outer Shell	Cylinder	Aluminum	0.132
			Head	"	0.073
	O_2	Tank	Cylinder	Inconel 718	0.185
			Head	"	0.093
		Outer Shell	Cylinder	Aluminum	0.104
			Head	"	0.073
Configuration B (1.4:1 Elliptical domes)	H_2	Tank	Cylinder	Inconel 718	0.055
			Head	"	0.036
		Outer Shell	Cylinder	Aluminum	0.148
			Head	"	0.090
	O_2	Tank	Cylinder	Inconel 718	0.185
			Head	"	0.121
		Outer Shell	Cylinder	Aluminum	0.119
			Head	"	0.090

in the multilayer insulation has been used. To determine the most suitable method of insulation of the AES vessels, the following methods were reviewed: (1) laminar wrap, (2) fiber glass, (3) perlite, (4) plating only, (5) plating with discrete shields, and (6) plating and vapor-cooled shields.

The pertinent factors to consider in the selection of the insulation method are the following: (1) heat leak, (2) vacuum retention shelf life, (3) simplicity, (4) reliability, (5) repeatability, (6) bake-out time and temperature, (7) ease of assembly, and (8) weight. These factors were considered and the result is in favor of the discrete shield concept with vapor cooling as required. The reasons are numerous but the overriding considerations are the simplicity and the ability to predict the heat-transfer performance.

A considerable amount of effort has been expended in the area of shield performance prediction for both discrete shields and vapor-cooled shields and for various temperatures and emissivities. A computer program was prepared to evaluate the effect of these parameters. The program calculates the heat leak of a dual-walled cryogenic storage vessel for conditions of radiation heat transfer only, on the basis of one square foot of parallel plates. The program evaluates multiple shielding combined with vapor cooling for any flow rates desired, and it evaluates the effect of electropolished and mechanically polished surface emissivities over a range of temperatures.

The results presented herein are divided into two groups, one for the oxygen and one for the hydrogen insulation performance. Only results for vapor-cooled shields are presented.

Figure 47 presents a summary of heat-leak performance per unit area as a function of oxygen flow rate per unit area for one shield, vapor-cooled. Further, this figure presents parameters of shield performance for both electropolished and mechanically polished shields for the complete range of temperature (80 F to 170 F). The curve at the top of each figure represents the maximum tolerable heat leak for constant pressure operation. This value increases with flow rate as shown. The oxygen curves show that a single vapor-cooled shield will yield a satisfactory vessel, especially if the total average temperature around the vessel is at or below 100 F, at which point even a poor electropolish job may be satisfactory.

The results of hydrogen tank insulation analyses are shown in Figure 48. The use of two shields, with only the inner shield vapor-cooled, results in a satisfactory heat leak at all temperatures shown if all surfaces are plated and electropolished. For this case, it may be possible to avoid

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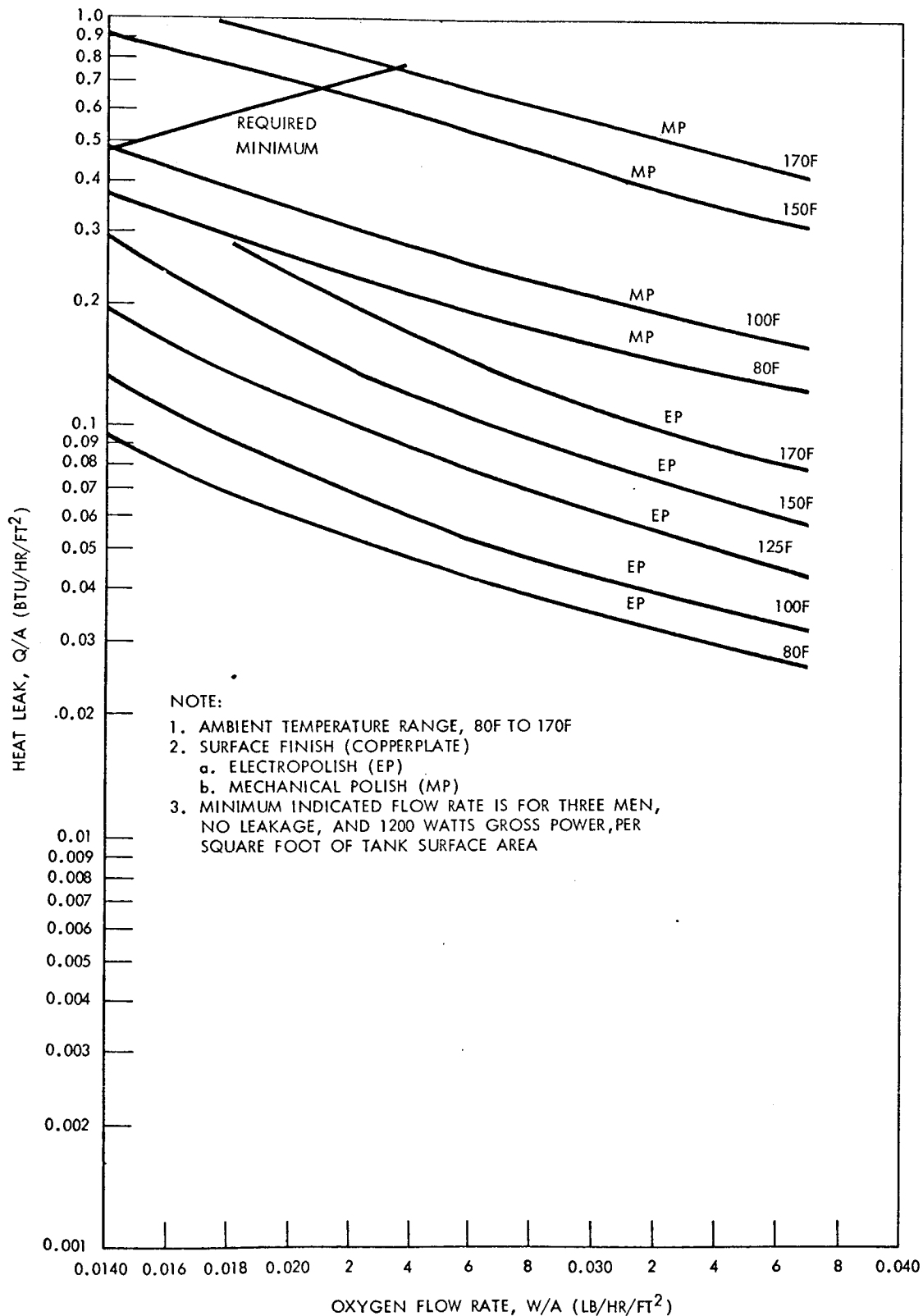
Figure 47. Oxygen Tank Radiation Heat Leak With One Vapor-Cooled Shield

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NOTE:
1. AMBIENT TEMPERATURE RANGE, 80F TO 170F
2. SURFACE FINISH (COPPER PLATE)
    a. ELECTROPOLISH (EP)
    b. MECHANICAL POLISH (MP)
3. MINIMUM INDICATED FLOW RATE IS FOR 1200 WATTS
    GROSS POWER, PER SQUARE FOOT OF TANK SURFACE AREA
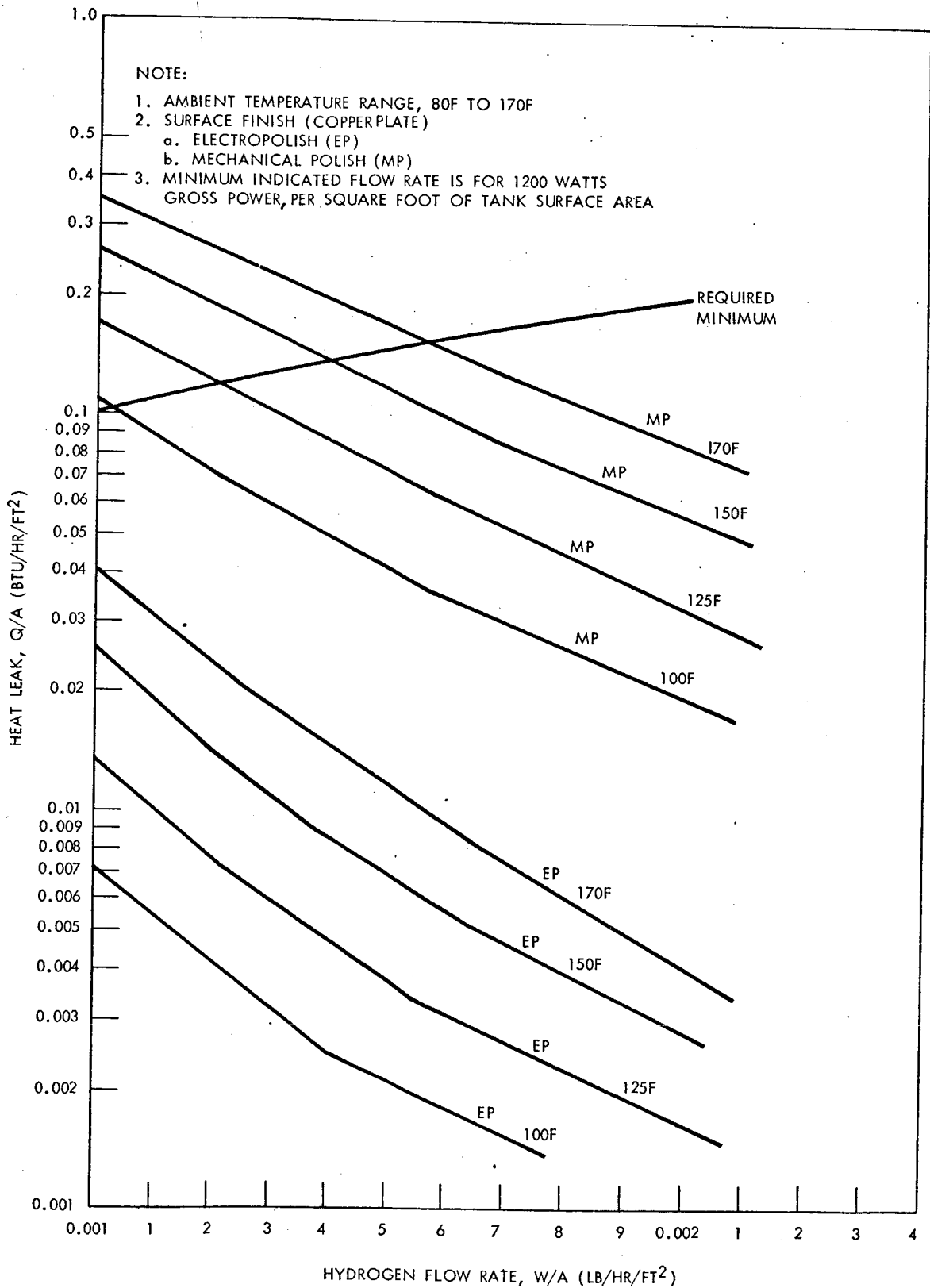
Figure 48. Hydrogen Tank Radiation Heat Leak With Two Shields,
One Vapor-Cooled

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plating the inner vessel and only plate and electropolish the shields and outer shell, which would result in decreased costs and fabrication time. The data show that none of the copperplated mechanical polish conditions are satisfactory. The case of two shields, both vapor-cooled, shows that nearly any condition will provide a low heat leak, especially when the ambient temperature is near or below 125 F.

AES CRYOGENIC STORAGE SYSTEM

A new cryogenic storage system was required for the AES missions. The oxygen and hydrogen storage tanks will be much larger than the Block II tanks and a new cabin atmosphere diluent tank system was required. The characteristics of this system are shown in Table 51. The schematic arrangement of the baseline cryogenic storage system is shown in Figure 49. The oxygen tanks are connected by plumbing and valves such that oxygen can be supplied to the environmental control subsystem and any of the four fuel cells from each tank. The hydrogen tanks are connected by plumbing and valves such that hydrogen can be supplied to any of the four fuel cells from each tank. The three nitrogen tanks are connected by plumbing to function as a single storage tank. The oxygen accumulator in the command module is assumed to be required, although the emergency flow requirement to the command module and laboratory in the event of a puncture is not stated as a functional requirement in the guidelines. The integration of the accumulator and the cryogenic storage system should be subject to further evaluation. Each fuel cell reactant shutoff valve package consists of two Block II reactant-shutoff valve packages that have been connected upstream of the check valves. New oxygen and hydrogen tank relief valves are required because of a higher flow due to larger surface area of the tanks. The pressure switches require further analysis with respect to reliability and heater power.

The nitrogen tanks are conventional cylindrical pressure vessels. The nitrogen fill valve, pressure regulator, pressure transducer, and relief valves are new, but a possibility exists that Block II SM RCS components can be used for this application. The alternate system (with cryogenic tanks in Sector IV only) is shown schematically in Figure 50. The cryogenic tanks and associated components are the same as for the baseline system.

A high-pressure ambient temperature abort storage system for oxygen and hydrogen located in SM Sectors II, III, IV, as shown in Figure 51 (drawing 5254-420), will be added for crew safety for lunar orbit missions, where the cryogenic tanks have been removed from Sector I. The high-pressure storage tanks are conventional cylindrical pressure vessels. Three vessels are required for hydrogen and two for oxygen. The three hydrogen tanks and two oxygen tanks are manifolded together, respectively, to function as a single storage tank for each fluid. As is the case for the nitrogen system, the fill valves, pressure regulators, pressure transducer, and shutoff valves are new, but a possibility exists that Block II SM RCS components can be used.

Table 51.   AES Cryogenic Storage System Characteristics

Type Fluid	Oxygen		Hydrogen	
Pressure Vessel Dome Configuration	Hemispherical	1.4:1 Ellipse	Hemispherical	1.4:1 Ellipse
Number of Tanks	2	2	2	2
Tank Volume (one tank)	18.8 ft^3	20.6 ft^3	28.1 ft^3	31.9 ft^3
Ullage and Residual Quantity	5 percent	5 percent	5 percent	5 percent
Usable Quantity (one tank)	1272 lb	1395 lb	118 lb	134 lb
Tank Dimensions – Diameter	41.5 in	41.5 in	41.5 in	41.5 in
– Length	43.3 in	41.9 in	56.4 in	57.8 in
Tank Weight (one tank)	339 lb	459 lb	270 lb	338 lb
Maximum Operating Pressure	1020 psia	1020 psia	300 psia	300 psia
Flow Rates (two tanks) – Maximum	12 lb/hr	12 lb/hr	1.0 lb/hr	1.0 lb/hr
– Minimum	1.0 lb/hr	1.0 lb/hr	.10 lb/hr	.10 lb/hr
Materials – Pressure Vessel	Inconel 718	Inconel 718	Inconel 718	Inconel 718
– Outer Shell	Aluminum	Aluminum	Aluminum	Aluminum
Construction and Fabrication Method Pressure Vessel	Forged and Machined	Forged and Machined	Forged and Machined	Forged and Machined
Outer Shell	Monocoque	Monocoque	Monocoque	Monocoque
Average Power Capability (2 tanks)(45 days)	2360 watts	2620 watts	2360 watts	2620 watts
Maximum Environmental Temperature	80 F	80 F	150 F	150 F
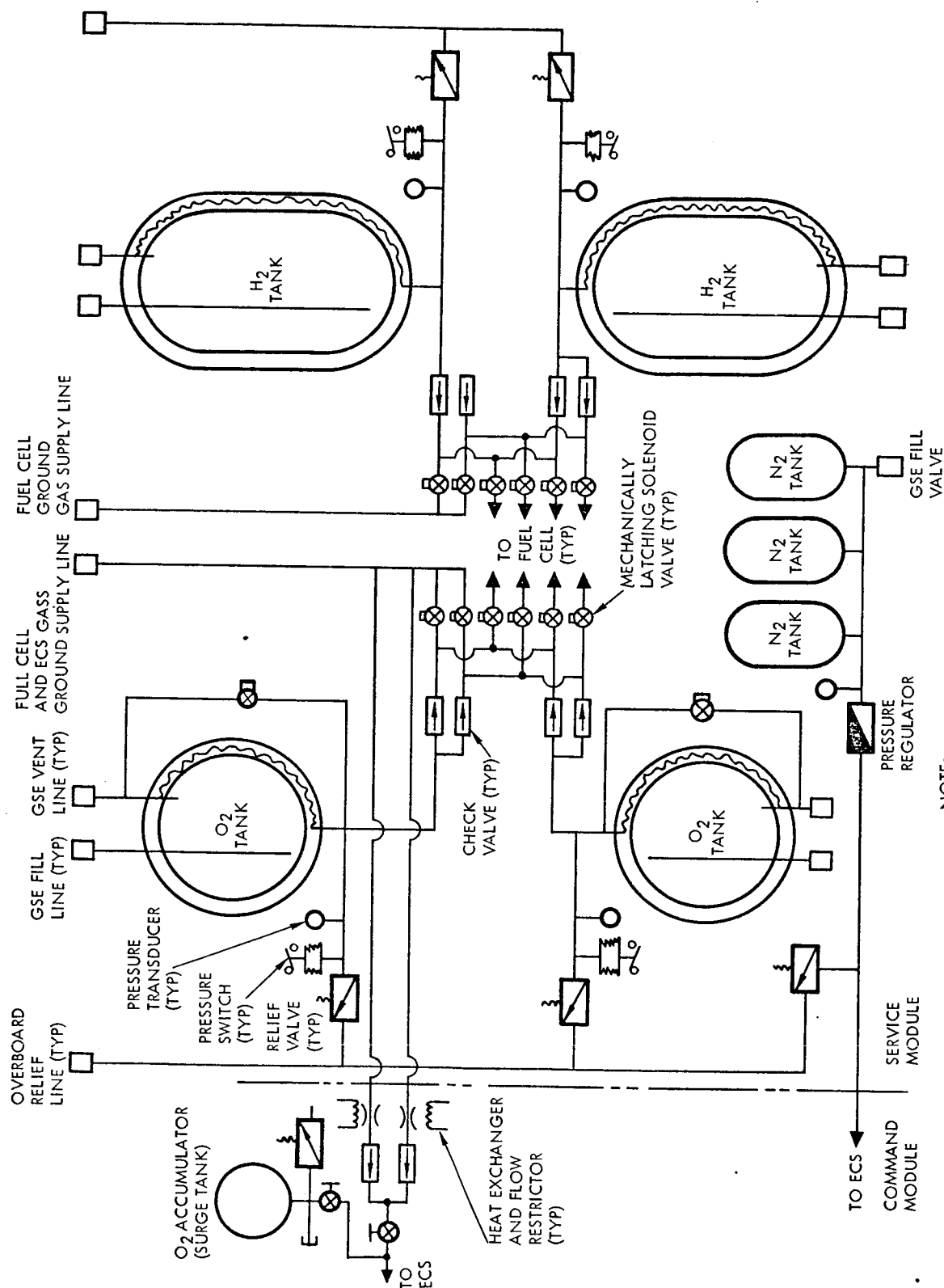
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Figure 49.   Schematic Arrangement of AES Cryogenic Storage Subsystem Baseline Configuration
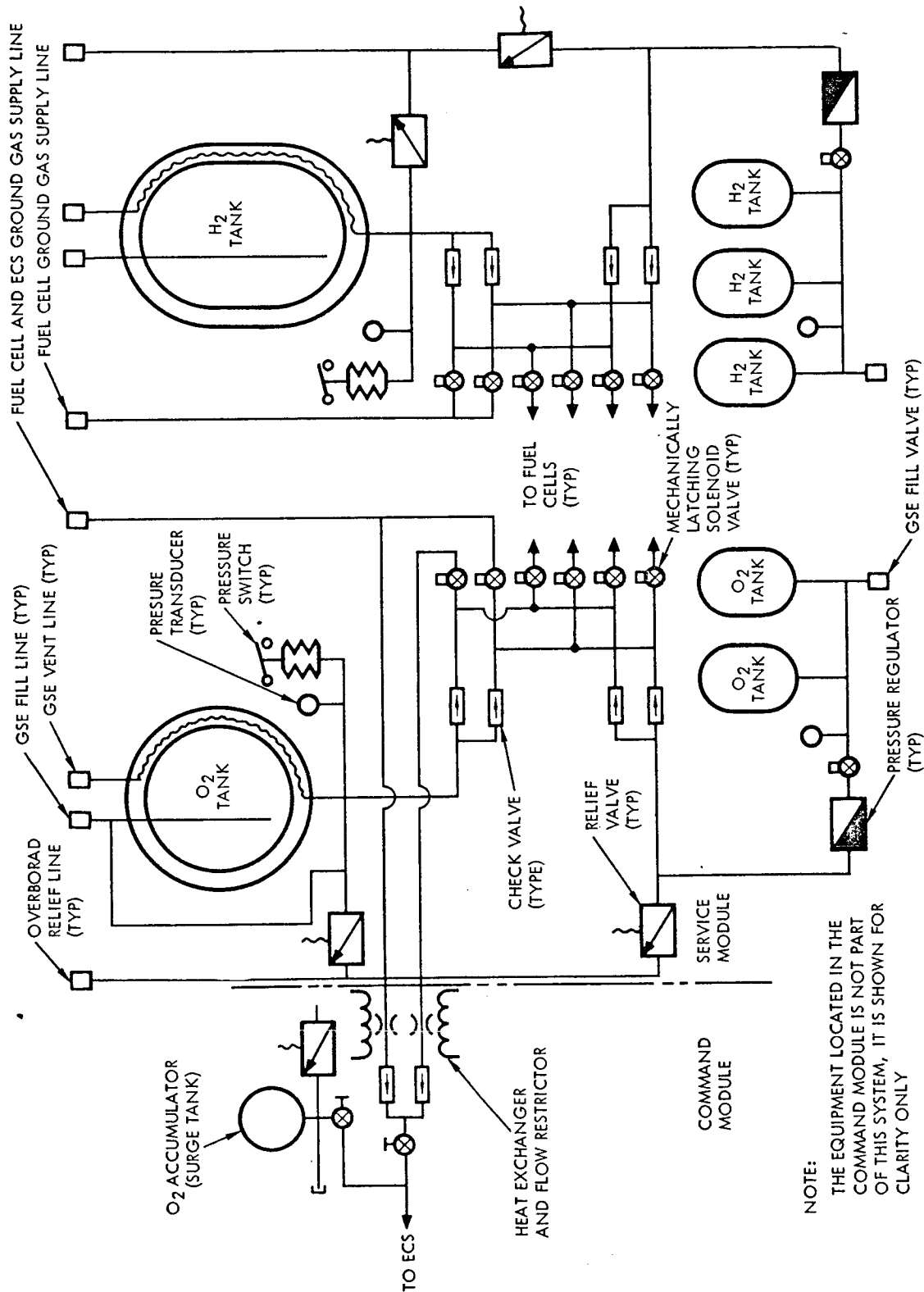
Figure 50.   Schematic Arrangement of AES Cryogenic Storage Subsystem Baseline Configuration with Pallet

VIEW LOOKING AFT

AFT

BAY I

-Z

RCS OXID

HE
HE
HE
HE

BAY VI

SPS FUEL

RCS FUEL

RCS FUEL

SPS FUEL

HE

HE

BAY V

HE
RCS OXID

-Y

OUTER ML CM
ABLATOR ML CM

X_s 355.00

BLOCK II

199.90

-Y

-Z

EPS RADIATOR (8 PLCS)
EPS RAD.
EPS RAD.

RED LT.

HELIUM
PYROTECHNICS
ACCESS

X_s 320.9

RCS OXID FILL & VENT
RCS FUEL FILL & VENT
RCS OXID FILL & VENT

X_s 268.50
BAY V X_s 255.34

ECS RADIATORS

X_s 213.00

R

BAY V

BAY VI

HE HE
HE HE
HE HE
HE HE

OXID
FUEL
FUEL
OXID

OXID
FUEL
FUEL
OXID

SPS FUEL
SPS FUEL

GAS O_2
GAS H_2
GAS N_2
GAS N_2

BAY V

BAY VI

-Z   FLY-AWAY UMBILICAL   **+Y**

EPS RAD.   EPS RAD.   EPS RAD. GREEN LT.   EPS RAD.   EPS RA
AMBER LT.

HELIUM PYROTECHNICS ACCESS

FUEL CELL FILL & VENT

RCS OXID FILL & VENT

H₂ PURGE

RCS OXID FILL & VENT

RCS FUEL FILL & VENT

H₂ FILL & VENT

O₂ FILL & VENT

ECS RADIATORS

O₂ PURGE

G   A

BAY VI   BAY I   BAY II   BAY

HE   HE   FUEL CELL 4   HE   HE   HE   HE   HE

OXID   OXID   FUEL   FUEL

OXID   CRYOGENIC H₂   OXID   FUEL   FUEL

SPS OXIDIZER   SPS

GAS N₂   CRYOGENIC O₂   GAS N₂   GAS H₂   GAS H₂

BAY VI   BAY I   BAY II   BA

+Z

C/M TO S/M UMBILICAL

EPS RAD.   EPS RAD.

X_S 376.00

AMBER LT.   AMBER LT

X_S 355.00

-Z

FUEL CELL
FILL & VENT

BAY VI

H_2 PURGE

RCS OXID
FILL & VENT

H_2 FILL &
VENT

BAY V

O_2 FILL &
VENT

-Y

O_2 PURGE   A

X_S 199.90

BAY III   BAY IV

X_S 355.00

HE   HE

FUEL
CELL   FUEL
CELL

HI-GAIN ANTENNA
(DEPLOYED)

OXID

1 2 3

BATT   BATT

OXID

CRYOGENIC H_2

14. GASEOUS HYDROGEN
13. GASEOUS OXYGEN
12. HEMISPHERE CRYO.
11. FUEL CELLS START
10. APOLLO BLOCK II A
9. DSIF ANTENNA -AP
8. INTEGRAL CRYOGEN
7. GASEOUS NITROGEN
6. FOUR FUEL CELLS
5. APOLLO BLOCK II EPS
4. APOLLO BLOCK II ECS
3. FULL SECTOR RCS
2. SPS PROPELLANT CA

NOTES: 1. RCS MOTORS ON Y
TANKS FOR PRESSU

GAS
O_2

CRYOGENIC O_2

X_S 199.90

BAY IV

Figure 51.   SM Final

VIEW LOOKING FORWARD

BAY I

HEAT SHIELD OUTER ML

-Z

+Y

BAY II

SPS ENGINE (NOT SHOWN FOR CLARITY)

BAY III

+Z

BAY IV

S/M TO SLA UMBILICAL
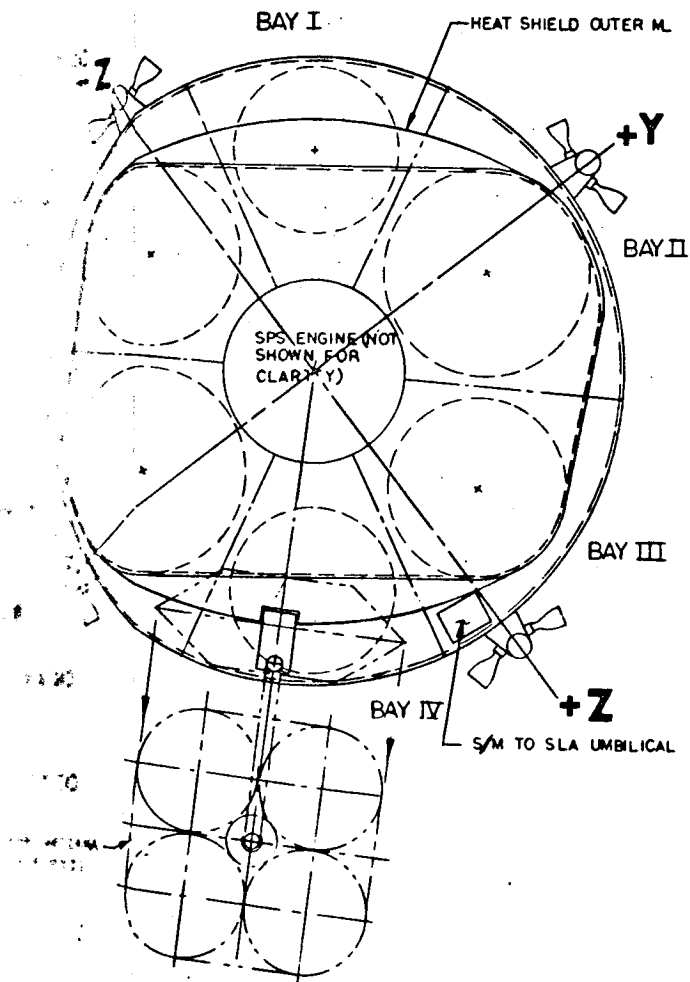
NITROGEN } USED, WHEN BAY I CRYO. REMOVED
OXYGEN

CRYO. TANK DOMES - TOTAL LENGTH 100.00 IN, DIAMETER 41.50 IN

START BATTERIES

BAY II AFT HEAT SHIELD MOD.

APOLLO BLOCK II RE-LOCATED

CRYOGENIC H₂O₂ IN BAYS I & IV

NITROGEN

CELLS

EPS RADIATORS

ECS RADIATORS

ECS DOOR PANEL (4 LEM OXIDIZER SIZE TANKS/SECTOR)

CAPACITY - APOLLO BLOCK II

Y & Z AXIS, APOLLO BLOCK II RCS HELIUM PRESURIZATION.

SCALE	BY	
1/20	DATE 11-12-1965	NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION 12214 LAKEWOOD BLVD. DOWNEY, CALIFORNIA
	MODEL	

S/M - FINAL BASE LINE, CONFIGURATION

5254-420

Final Baseline Configuration

- 241, 242 -

POWER-GENERATING CAPABILITY

The average electrical power levels that can be generated for 34-day missions and 45-day missions with the two selected tank configurations are shown in Figures 52 and 53. Both the baseline spacecraft configuration and the "Sector IV only" case are shown.

The 34-day missions with single cryogenic storage tanks are significant in that the addition of the laboratory with its attendant oxygen leakage rate causes a significant decrease of available power and also causes the missions to be shortened; however, in each case the remaining hydrogen available for power generation is adequate to support two fuel cells at minimum power (1600 watts) for 34 days. Therefore, a feasible approach to increase the mission duration to 34 days is to include an additional quantity of oxygen for the laboratory leakage. The additional oxygen can be provided by several methods: (1) increase the oxygen tank volume by penetration of the SM aft bulkhead and retain the hemispherical domes on the pressure vessels, (2) increase the oxygen tank volume and use elliptical domes on the pressure vessels such that penetration of the aft bulkhead is not required, (3) add a small oxygen tank in one of the SPS fuel or oxidizer bays (a second identical tank could be used for nitrogen if a cryogenic nitrogen storage is selected), and (4) add a Block II oxygen tank in the laboratory and transfer oxygen from the laboratory to the CSM ECS.

The use of elliptical domes on the pressure vessels without penetration of the aft bulkhead and the addition of a Block II oxygen tank in the laboratory appears to be the most attractive solution.

The study also showed that by increasing the elliptical ratio to approximately 2.25:1, and without penetration of the aft bulkhead, all environmental oxygen and all electrical power reactants (1930 watts) for the CSM and the laboratory for the 34-day lunar mission can be stored in Sector IV.

It appears feasible to supply all electrical power for the 34-day mission from the CSM if a Block II oxygen tank is added in the laboratory. However, detailed electrical load analysis of this mission must be completed to verify that the reactant quantities will be sufficient.

If additional power is required, a 6.5 percent increase can be obtained by using a high-density fill method (subcooling). Power increases beyond that level can be obtained by bulkhead penetration or increased tank diameter or a combination of both.
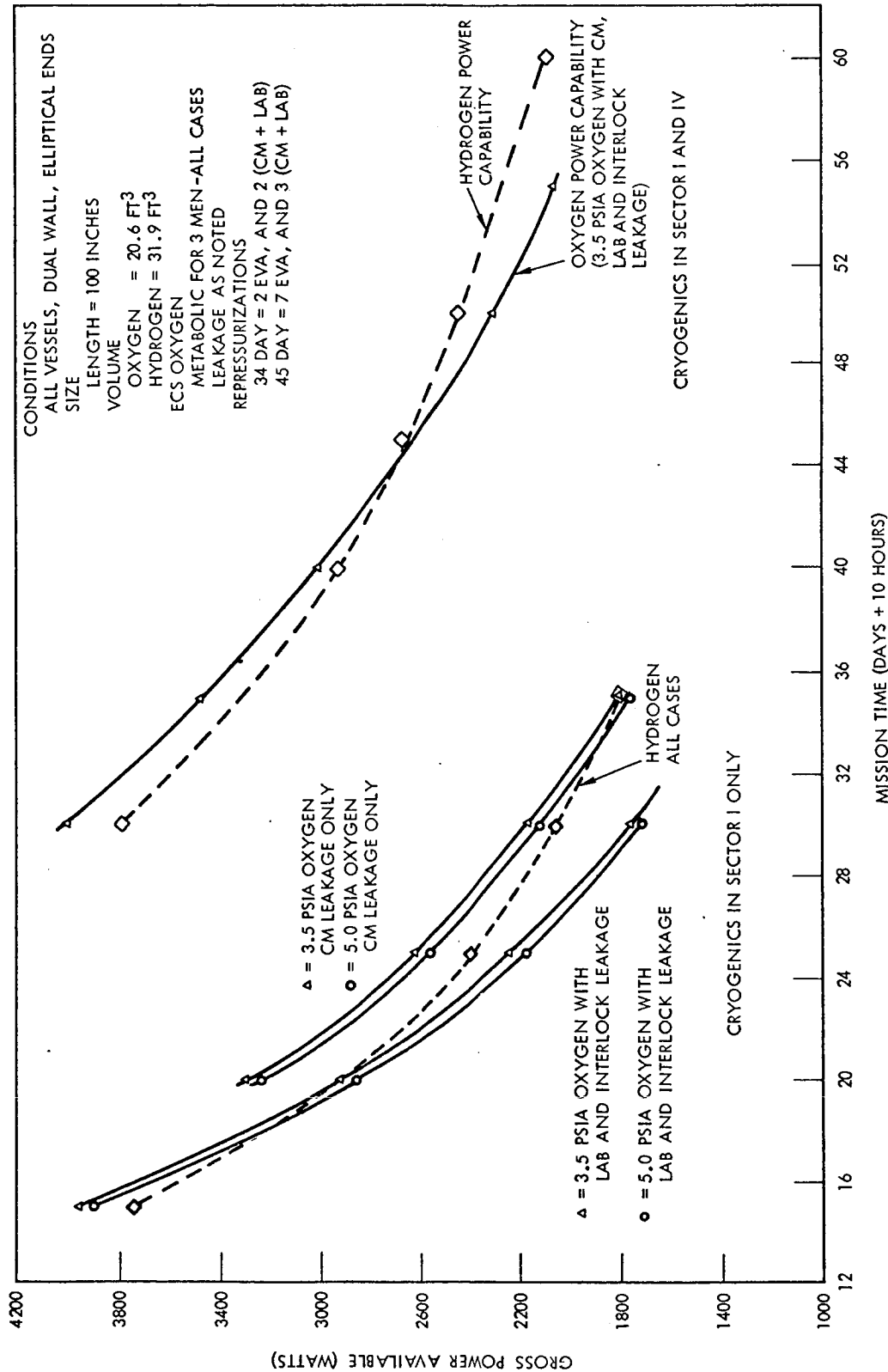
Figure 52.   Mission Capability, Elliptical Domes
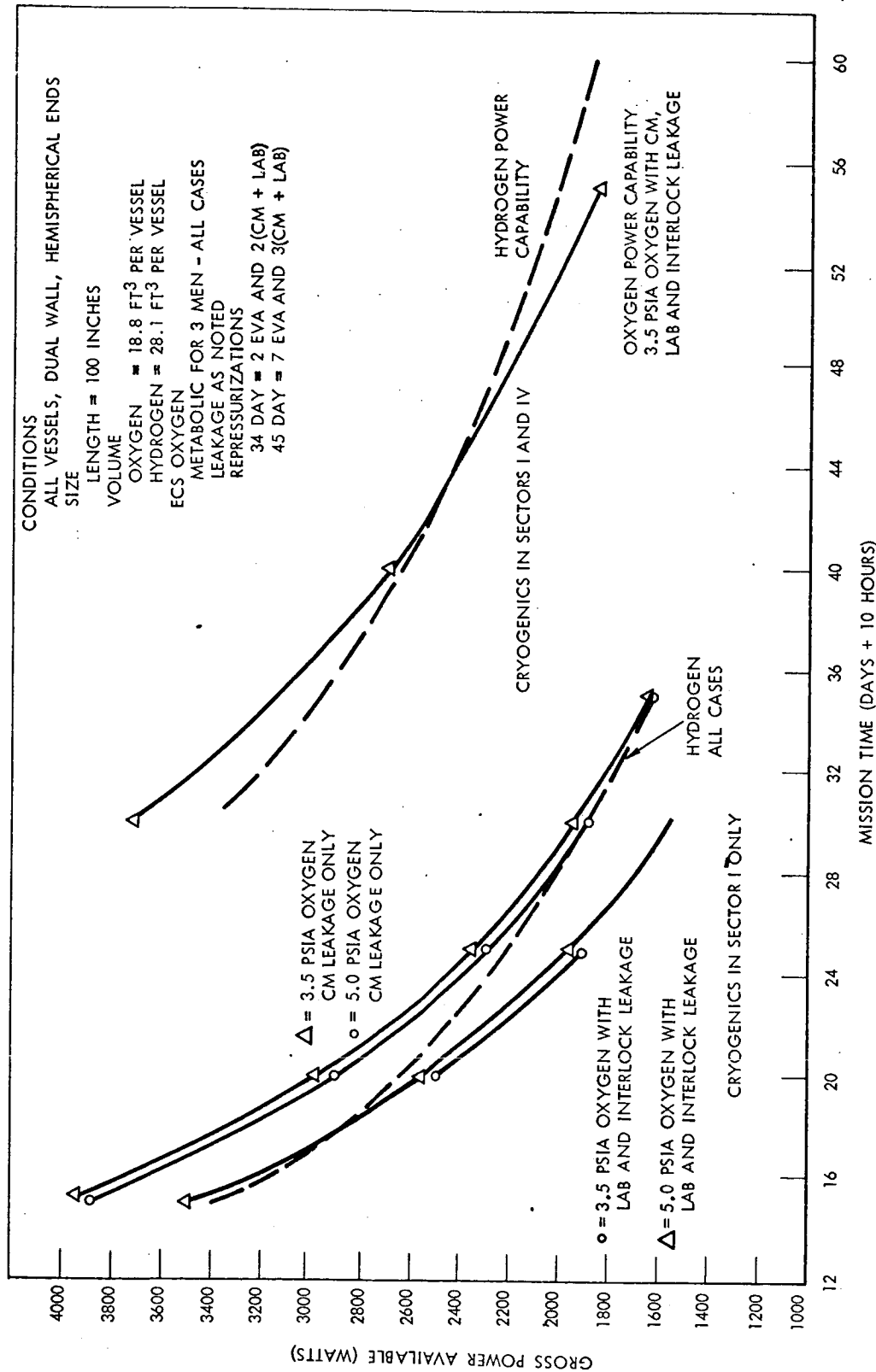
Figure 53. Mission Capability, Hemispherical Domes

CONCLUSIONS AND RECOMMENDATIONS

The principal objective of the study was to develop criteria for sizing the AES cryogenic storage system. That tank size which conforms to all of the ground rules is referred to in this report as the baseline system with the tanks sized for a balanced ratio of oxygen and hydrogen for the 45-day reference missions. The baseline system configuration includes cryogenic tanks in SM Sectors I and IV and three cylindrical nitrogen high-pressure tanks located in Sectors II and VI. The baseline-size tanks will support both the CSM and laboratory electrical and metabolic and leakage requirements, as presently known, for the 45-day missions.

Larger tank volumes than the baseline can be obtained by several methods: (1) penetration of the SM aft bulkhead, (2) increase the tank diameter by relocating the horizontal stiffeners on the radial beams in Sectors I and IV, and (3) use of elliptical domes on the pressure vessels.

The study showed that a larger increase could be obtained by penetration of the aft bulkhead than by increasing the tank diameter. It was also determined that the same tank volume that can be obtained by maximum penetration of the aft bulkhead with hemispherical domes can also be obtained by using elliptical domes on the pressure vessels without penetration of the aft bulkhead but with a weight penalty. This weight penalty is, however, largely eliminated if the subcritical storage method is used.

The alternate configuration (with cryogenic tanks in Sector IV only) presents a problem in that the baseline size cannot supply all of the environmental oxygen for the CSM and the LEM laboratory plus the reactants for two fuel cells for the 34-day reference lunar mission. It appears possible to supply all electrical power required for the 34-day mission from the CSM if the elliptical configuration is selected and a Block II oxygen tank is added in the laboratory.

The recommended design concept which includes vapor-cooled shields and local supports appears to be ideally suited for subcritical storage of oxygen and hydrogen for AES. Because of the lack of good flight data on subcritical storage, it is recommended that the baseline approach be supercritical storage with subcritical storage as an alternate. It is recommended that the boilerplate tank test program be performed with both methods (both supercritical and subcritical tests can be performed with the same vessel) and that a final selection be made based on the results from these tests and the results of the pending flight test to be conducted by NASA.

SID 65-1519

SERVICE PROPULSION SYSTEM

SERVICE PROPULSION SYSTEM

This section presents a summary of the AES service propulsion system (SPS) analyses performed during the preliminary definition phase. It includes a statement of AES requirements for the system based on four representative reference missions, a preliminary evaluation of the capability of the Block II SPS to meet these requirements, and a description of the system design changes required. It was assumed that the Block II system will have been fully qualified for the performance and environmental requirements of Apollo 14-day earth orbital and 10-day lunar missions.

The study approach consisted of comparing the AES and Apollo mission profiles and propulsion performance requirements to define differences. The most significant difference is the AES increase in mission duration to 45 days. The study of long-term, space-environmental-exposure effects and the definition of system changes for coast life extension was emphasized. These factors involved both materials life in the space environment and a reassessment of SPS propellant and helium fluid leakage implications.

The comparison of performance requirements revealed no differences between the AES and Apollo missions except that a large number of AES earth-orbit missions requires only a fraction of the Block II propellant capacity. Since a study ground rule required that potential weight savings be identified, a trade-off study of several propellant tankage configurations was conducted and resulted in a recommended alternate design to be used on these missions.

For a more detailed description of the studies conducted on the SPS the reader is referred to NAA/S&ID report SID 65-1527.

AES REQUIREMENTS

The requirements imposed on the service propulsion system by the AES program are derived from the characteristics of four representative AES reference missions. A detailed description of these reference missions is contained in the Systems Analysis report, SID 65-1534. These missions were used as the basis for all analyses and trade-offs conducted during this study. The requirements of these reference missions make it mandatory to retain the Block II SPS tankage as the basic AES configuration. Several

other missions, which have been defined only in a preliminary manner, require considerably less SPS propellant than the Block II and logically lead to an alternate selection of a reduced propellant tankage configuration.

Maximum duration of the reference missions is 45 days, whereas the Block II requirement is a maximum of 14 days. The longer duration places more stringent requirements on the SPS. The longer exposure to propellants and combined environments is an area that requires extensive analysis and eventually demonstrated capability by test.

PERFORMANCE REQUIREMENTS

With respect to primary performance characteristics, AES missions have been designed to fall within the anticipated maximum propulsion capability of the Block II SPS. These primary characteristics are shown in Table 52.

PROPELLANT REQUIREMENTS

The AES usable propellant requirements for each flight were estimated (see report SID 65-1547, Performance Analysis, Phase II Flights) and are presented in Figure 54.

The Block II maximum usable propellant capability of 39,700 pounds is required for some flights. This slightly exceeds the nominal requirement of

Table 52. System Requirements

Item	Requirement
Reliability Goal (LPO Mission)	
Mission Success	0.99868
Crew Safety	0.9999
Dry Weight (including tanks) (lb)	2,830
Usable Propellant (maximum) (lb)	39,700
Thrust (nominal) (lb)	20,000
Specific Impulse (3-sigma minimum) (sec)	311.7
Minimum Impulse per Start (lb sec)	5000±200
Shutdown Impulse (lb sec)	8830 to 14,200±300
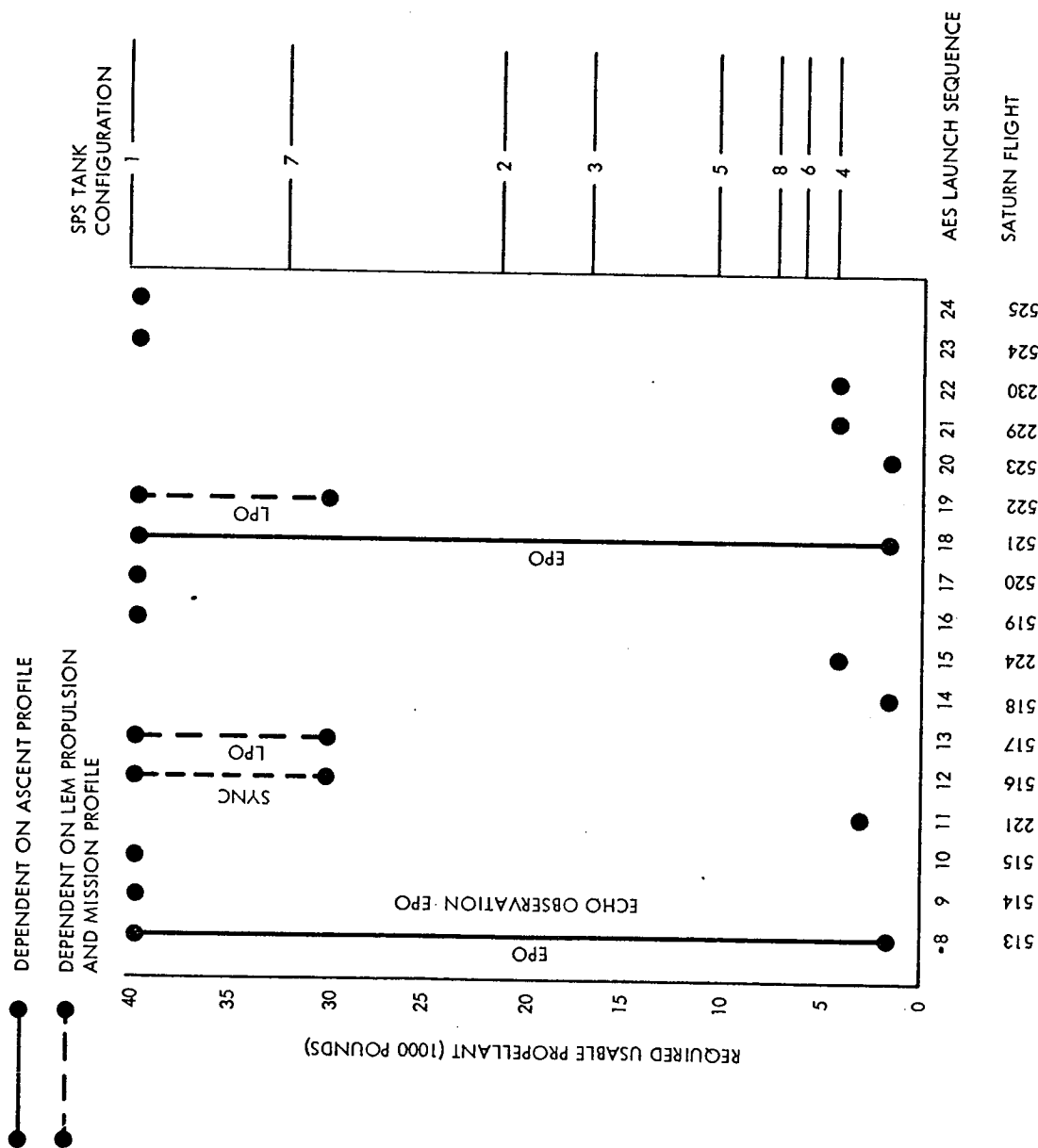
Figure 54. AES Phase II Mission SPS Propellant Requirements

38,000 pounds for the Block II LOR mission. On six other flights, notably low-inclination, earth-orbital missions, the propellant requirement is less than 10,000 pounds.

SPECIFIC IMPULSE

The AES specific impulse requirement is constrained to the Block II value (see report SID 65-1527) although higher performance would increase AES CSM utility.

THRUST

The nominal thrust predicted for the Block II SPS is 20,000 pounds. The AES requirement is constrained to this value. Additional thrust is of no advantage except for suborbital abort and flights using SPS for earth-orbital injection where an increase in payload would be possible.

DUTY CYCLE

The maximum number of in-flight starts required for AES missions considered to date does not exceed 15. For these missions, the minimum coast time between long burns is not less than two hours. Neither of these duty-cycle characteristics is a firm requirement at this time, however, since AES missions are not yet final. Therefore, SPS capability should permit flexibility in this area. The maximum coast period between firings is 45 days.

MINIMUM IMPULSE AND REPEATABILITY

The AES requirements for SPS minimum impulse, shutdown impulse, and impulse repeatability have been assumed to be identical with the Block II requirement. This is subject to verification during guidance and control requirements definition in future studies.

REDUCED PROPELLANT TANKAGE STUDIES

Several AES missions require considerably less than the maximum propellant load and can be benefited by the weight and volume savings attainable by eliminating excess tankage on these flights instead of merely off-loading propellant. The weight saving attained increases with the reduction in capacity of the configuration selected. However, the complexity and cost of the changes required varies widely over this range. For this reason the study investigated discrete design points of the curve in detail. Configurations studied required only available hardware and minimum related changes and yet covered a wide variety of propellant loadings.

A total of eight tank configurations were considered that utilize hardware available from the Apollo and LEM programs. Of these, the simplest configuration (involving only the removal of the propellant storage tanks and related minor changes to gauging and plumbing) was recommended. This configuration has a capacity for 53 percent of the maximum usable propellant load and saves 586 pounds of burned weight with minimal change cost.

INITIAL TRADE STUDY

The eight configurations covered in the initial study are shown in Figure 55 and correspond to the cases shown in Figure 54. Table 53 contains detailed descriptions of each case and the associated changes required. The table also shows the results of evaluation of the factors involved: weight and volume savings, usable propellant, cost, reliability, and test program impact. Propellant tankage, helium vessel, and tank internal descriptions are also shown. Tank system weight and volume savings, as given in Table 53, are approximately represented in Figure 56. A brief notation of required hardware changes is also included so that system change complexity is identified.

The c. g. limits for each configuration are included in the matrix but did not enter into the trade-off evaluation since all of the configurations fall within the SPS gimbal capability. The reliability estimates shown are based on the best information presently available, but are approximate because of Block II reliability data on qualified Block II or LEM hardware is not available. GSE changes are considered minor and do not materially affect the trade-off evaluation. All of the configurations, including the Block II baseline configuration, must be partially requalified for the 45-day AES environment, but LEM hardware and all Apollo hardware that is modified must be requalified for functional certification.

The recurring cost saving is approximately the same for each case estimated, while the nonrecurring costs vary considerably. Case 2 costs about $1100 per pound of weight saved while Cases 4, 5, and 6 cost between $2400 and $2800 per pound of weight saved.

TANK SIZING METHOD

The usable propellant capacities and the helium requirements for the eight AES SPS tankage configurations are shown in Table 53. Except for Case 1, which is the reference design, the propellant capacities and helium requirements were obtained by design calculations. Apollo Block II data were used where applicable and the results are considered approximate but conservative.

The unpressurized volume of the propellant tank under consideration was calculated if this information was unavailable. Calculations of residual volumes included: empty tank vapor, retention reservoir inside pull-through, retention reservoir outside pull-through, gauging system tolerances, and GSE loading tolerances. Residual volumes and estimated minimum ullage were subtracted from the total tank volume. The tank volume remaining was considered usable propellant volume.

The helium requirement for each configuration was obtained from an iterative end-point analysis. The helium required to displace propellant was calculated considering the volume of the residuals, tank expansion due to pressure and temperature, and helium compressibility. For this study, the method of calculation used was considered satisfactory since the trade-off study was of a preliminary design nature and the results are considered conservative.

GAUGING SYSTEM CHANGES

A trade-off study of propellant quantity gauging system changes for the reduced tankage configuration was conducted. Tankage configurations considered were Case 2, and the shortened propellant tank Cases 5, 6, and 7.

Case 2 tankage configuration deletes two 45-inch storage tanks and the corresponding gauging sensors. Deletion of the sensors affords a weight reduction of 21.5 pounds. Power consumption will also be reduced by about 0.5 watt for the two sensor removals.

The simplest change to the control unit is to utilize the existing unit with a sensor simulator to replace the deleted sensors. The net weight reduction would then be 19.75 pounds and is recommended for Configuration 2. Shortened propellant tanks, Cases 5, 6, and 7, require more extensive changes and testing, and would probably require a new control unit.

CASE 3A

OXIDIZER BAY III
FUEL   BAY IV

45 IN. DIA

109.7 IN.

51 IN. DIA

102.8 IN.
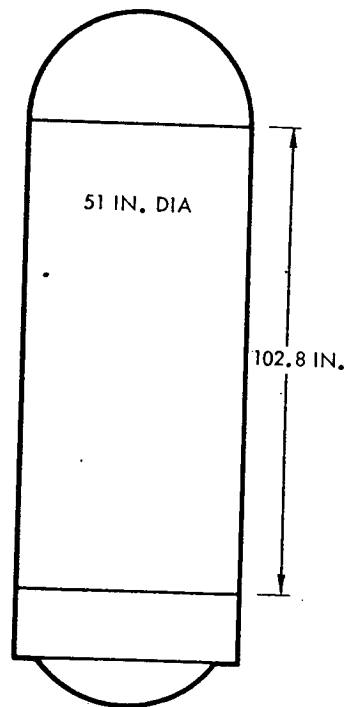
CASE 2

OXIDIZER BAY II
FUEL   BAY V
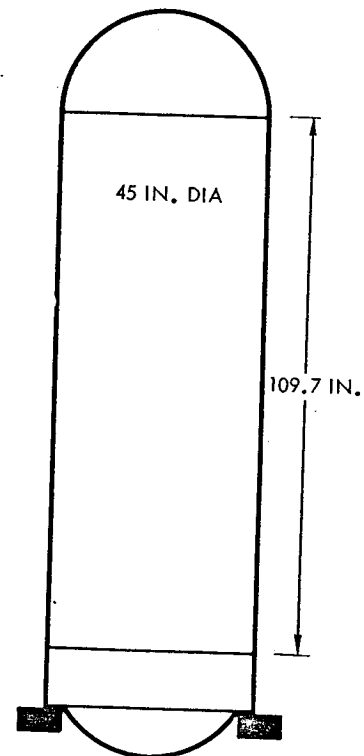
CASE 3B

OXIDIZER BAY II
FUEL   BAY V
ADAPTER REQUIRED

OX:
FUE:
AD:

SM BAYS LOOKING FORWARD

NOTE:

FUEL AND OXIDIZER
TANKS ARE EQUAL SIZE

49.4 IN.
SPHERE
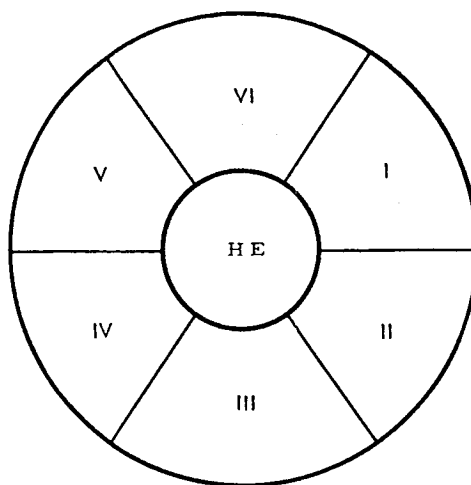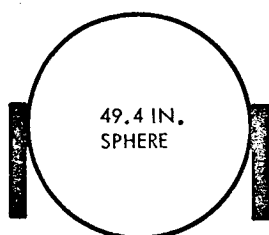
CASE 4

OXIDIZER BAY II
FUEL   BAY V
ADAPTER REQUIRED

51 IN. DIA

35.8 IN.
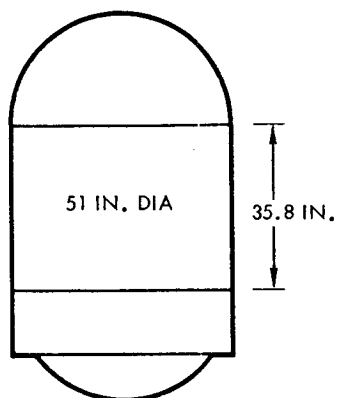
CASE 5

OXIDIZER BAY I!
FUEL   BAY V

51 IN. DIA

9 IN.

CASE 6

OXIDIZER BAY II
FUEL   BAY V

45 IN. DIA

109.7 IN.

51 IN. DIA

53.9 IN.

51 IN. DIA   19.29 IN.

9 IN.

CASE 7

OXIDIZER BAY III & II
FUEL   BAY VI & V

CASE 8
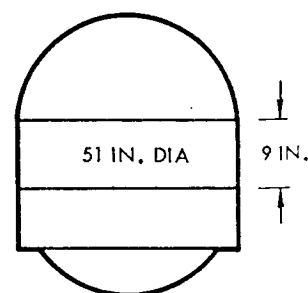
OXIDIZER BAY II
FUEL   BAY V
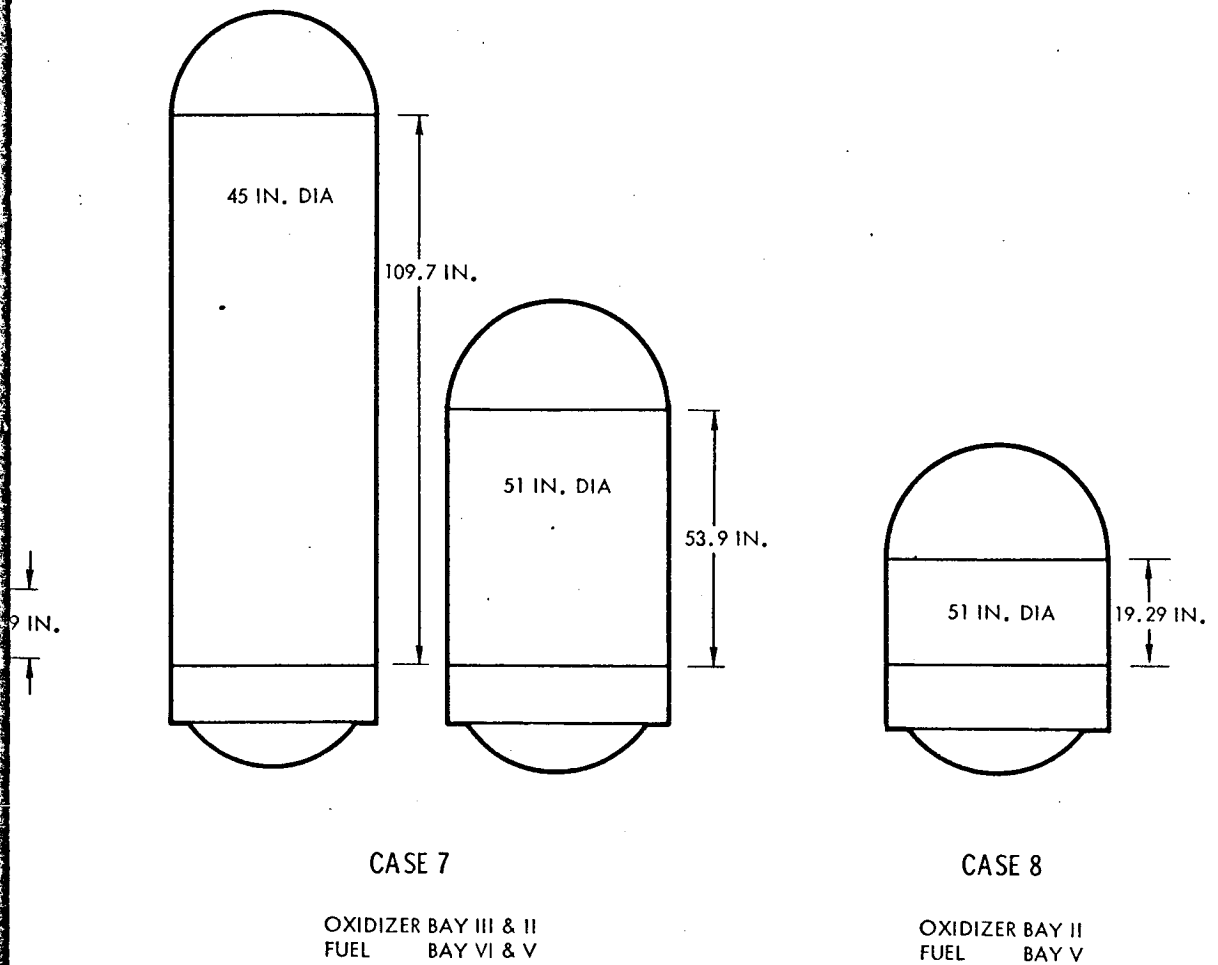
Figure 55.   Reduced Tankage Configurations

- 255, 256 -

Tank Configuration	1	2
Description	Block II	51-inch sumps
Useable Propellant MR 1.6 (Pounds)	38,000 (nominal)	21,040
*Residual Propellant	543	435
Storage Tank		
Type	Block II	None
Diameter (Inches)	45	—
Cylinder Length (Inches)		—
Oxidizer Location (Section)	III	—
Fuel Location (Sector)	VI	—
Sump Tank		
Type	Block II	Block II
Diameter (Inches)	51	51
Cylinder Length (Inches)	102.8	102.8
Oxidizer Location (Sector)	II	II
Fuel Location (Sector)	V	V
Helium Tanks		
Number and Type	Two Block II	One Block I
Diameter (Inches)	40.7 OD	40.7 OD
Fill Pressure (PSIG)	3585	3920
Qualified Pressure	3685	4000
Helium Weight (Pounds)	88.5	47.0
Tank Internals		
Retention Reservoir	Block II 51-inch, 58-inch height	Block II 58-inch height
Retention Screens	Block II 51-inch	Block II
Gaging Screens	Block II	Block II sump omit storage
Tank Door	Block II	Block II
Tank System Weight Saving Including Residuals (Pounds)	None	864
Volume Saving	None	Sectors III and VI
Hardware Changes	None	Remove 45-inch sto~ Reroute helium plum~ Remove one helium ~ Adapt gauging contr~
C.G. Limits		
X-Y Plane	-0.5 to 2.0°	0.1 to 2.85°
X-Z Plane	0.6 to 3.0°	0 to 1.9°
Reliability of Tanks (45 Days) Failures per 10^6 Missions	204	102
CSE and Ground Operations Changes	None	Minor ACE modifica~ for PUGS
Development Testing and/or Requalification	45-day environment	45-day environment Requalify helium ta~
Nonrecurring Δ Price (dollars)		969,000
Recurring Δ Price/Unit (dollars)		-213,000

*Includes all structure, propellants, residuals, engines, and lines, etc
Loading tolerance not included.

Table 53. Tank Trade-off Matrix

3A	3B	4	5	6	7	8
45-inch sumps Sectors III and VI	45-inch sumps Sectors II and V	Two LEM ascent tanks	10,000-pound 51-inch sumps	Minimum tank 51-inch sumps	80% Block II Reduced 51-inch sumps	Two LEM descent tanks
16,550	16,550	4270	10,000 (nominal)	5830	30,400	7500
419	419	393	405	397	527	400
None — — —	None — — —	None — — —	None — — —	None — — —	Block II 45 III VI	None — — —
Block II storage 45 109.7 III VI	Block II storage 45 109.7 II V	LEM ascent 49.4 ID Zero II V	Block II shortened 51 35.8 II V	Block II shortened 51 9.0 II V	Block II shortened 51 53.9 II V	LEM descent 51 19.29 II V
One Block II 40.7 OD 3200 3685 39.4	One Block II 40.7 OD 3200 3685 39.4	Two LEM ascent 22 OD 3500 3500 11.6	One LEM descent 33.3 ID 3400 3500 23.7	One LEM descent 33.3 ID 2400 3500 17.4	Two Block II 40.7 OD 2740 3685 67.9	One LEM descent 33.3 ID 2800 3500 20
Block II 58-inch height Block I fuel or adapt Block II Change Block II sump Omit Block II storage Block II	Block II 58-inch height Block I fuel or adapt Block II Change Block II sump Omit Block II storage Block II	Block II shortened Block II adapted Block II sump adapted LEM adapted	Block II (may be shortened) Block II Block II shortened Block II	Block II shortened Block II Block II shortened Block II	Block II shortened Block II Block II shortened Block II	Block II shortened Block II Block II shortened Block II
1057	1067	1545	1236	1341	246	
Sectors II and V	Sectors IV and VI	Sectors III and VI and Δ of 105 inches in Sectors II and V	Sectors III and VI and Δ of 67 inches in Sectors II and V	Sectors III and VI and Δ of 93.8 inches in Sectors II and V	Δof 48.9 inches in Sectors II and V	Sectors III and VI and Δ of 83.5 inches in Sectors II and V
Remove 51-inch sumps Reroute all plumbing Adapt internals Remove one helium bottle	Remove 51-inch sumps Move 45-inch tanks Add tank supports Reroute helium plumbing Adapt internals Remove one helium bottle	Remove Block II tanks Add two LEM ascent tanks Add tank supports Reroute helium plumbing Adapt internals Remove Block II helium bottles Add two LEM ascent bottles	Remove 45-inch storage Shorten 51-inch tanks Reroute helium plumbing Remove helium bottles Add LEM descent helium bottle Adapt gauging system	Remove 45-inch storage Shorten 51-inch tanks Adapt internals Remove helium bottles Add LEM descent helium bottle	Shorten 51-inch sumps Adapt gauging system	Remove Block II tanks Add two LEM descent tanks Adapt tank supports Reroute plumbing Adapt internals Remove Block II helium bottles Add two LEM descent bottles
0.1 to 0.8° 0.2 to 3.8°	0.1 to 2.7° 0.1 to 1.9°	0.1 to 1.0° 0.2 to 1.8°	0.2 to 2.0° 0.1 to 1.9°	0.1 to 1.3° 0.1 to 1.9°	Not Determined	Not Determined
102	102	Not known	1729 with LEM He 102 with Apollo He	1729 with LEM He 102 with Apollo He	204	Not known
45-day environment Major ground-test verification firings	45-day environment	45-day environment + LEM tank requal + internal requal	45-day environment + short tank requal + internal requal	45-day environment + short tank requal + internal requal	45-day environment + short tank requal + internal requal	45-day environment + LEM tank requal + internal requal
Not estimated	Not estimated	4,278,000	3,024,000	3,220,000	Not estimated	Not estimated
Not estimated.	Not estimated	-220,000	-244,000	-244,000	Not estimated	Not estimated
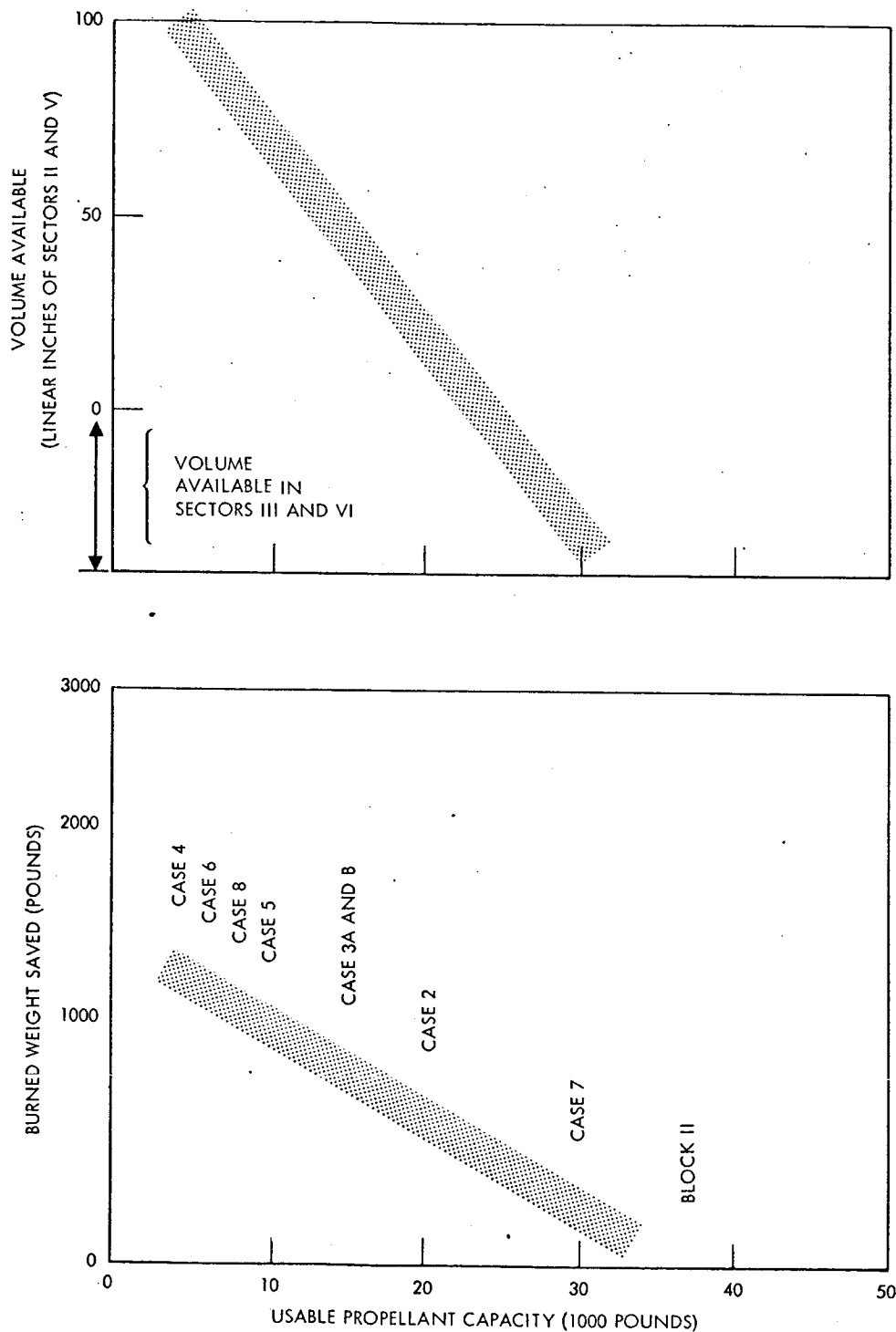
Figure 5.6. Weight and Volume Saving Versus Usable Propellant Capacity

CONFIDENTIAL

FLUID LEAKAGE

Extension of the Block II SPS to 45-day capability requires reassessment of fluid leakage from the system. The fluids considered are: (1) nitrogen, used for engine valve actuation, (2) helium, used for propellant pressurization, and (3) propellants. The types of leaks considered include: (1) internal leaks, causing pressures to exceed design limits or causing materials compatibility and/or explosion problems, and (2) external leaks, depleting gas or propellant. Fluid leakage is considered from three viewpoints: (1) the problem of predicting in-flight leak rates (assuming no failures) over long periods from preflight acceptance tests, (2) a review of Block II acceptance test leak rates for adequacy on an extended mission, and (3) the measurement of fluid leakage continuously throughout a mission to permit timely abort action.

The Block II design philosophy used to establish standards for acceptance test leak rates has assumed that the increase in leak rate during a mission would not be excessive unless a failure with an assignable cause occurred. Moreover, acceptance test methods for liquids (propellants) utilize a gas substitute to improve accuracy in leak-rate measurement. This same philosophy is used for a reassessment of Block II leak rates for AES 45-day missions. However, it is apparent that for longer duration missions the prediction of in-flight leak rates on the basis of a short-term acceptance test leaves room for considerable doubt. Progressive degradation in the performance of seals is a well known and common occurrence, usually compensated for by maintenance. Such deterioration is not generally covered as a failure by reliability assessment. In any event, an improvement in predicting long-term performance of individual seals (which can vary grossly from that of qualification test articles) on the basis of prelaunch acceptance tests is required for the AES SPS. The problem primarily involves test technique and will be emphasized in future studies.

The study results indicate two potential problem areas that require additional analysis. These are: (1) propellant backflow leakage across the helium check valve and into the helium regulator region and (2) propellant leakage from the shaft seals into the common gear housing of the propellant valve acutation system.

BACKFLOW LEAKAGE TO REGULATOR

A potential problem exists for the AES mission since a corrosive or explosive mixture of helium and propellants may be attained above the check

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valves and in the region of the regulators. This plumbing is common to both fuel and oxidizer backflow above the check valves. There are three possible ways that propellant, liquid or vapor, might reach the check valve region:

1. When the helium solenoid valve is closed, helium leaking externally from the region above the check valve will cause a differential pressure that, with backflow helium leakage through the check valve, will allow a slug of propellant to move from the propellant tank towards the check valves.

2. Even without an external helium leak, the propellant vapor will diffuse into the helium and eventually this diffusion will reach the check valve region.

3. In zero gravity, films of propellant will climb the helium pressurization lines and may reach the check valves.

Only the first method is considered a problem for the AES mission. The time required for diffusion is of the order of 1500 hours before propellant vapor would reach the check valve region. The phenomenon of films of propellant climbing the helium lines in zero gravity is line length dependent and it is doubtful that propellant would get to the check valves.

The AES mission is approximately 1000 hours and the specification external leak rate of the four check valves is 2.0×10^{-4} scc/sec of helium. Considering the volume of the helium system from the check valves to the top of the standpipe, a slug of propellant could conceivably reach the check valves due to the driving pressure differential after 500 hours of coast. Additional external helium leakage above the check valves will cause a pressure differential that, with backflow leakage across the check valves, will allow fuel and oxidizer to reach and mix in the common regulator region.

MAIN PROPELLANT VALVE SHAFT SEALS

A potential explosive or corrosive problem is a possibility with the Block II Apollo configuration of the SPS engine ball valve shaft seal drainage system. The ball valve shaft seals have a specification leak rate maximum of 40 scc/hr of GN_2. This rate of GN_2 is equivalent to 2 cc/hr of propellant. Under these conditions, sufficient propellant leakage can result to permit the propellant to freeze in the unheated drain line as a result of free expansion to the vacuum environment. If enough propellant freezes in the drain line, blockage will occur. Propellant trapped in the blocked drain line can then either leak into the common gear compartment and cause a possible corrosive or explosive condition or build up pressure that will impair the integrity of

the valve shaft seal. This problem is common with the Apollo mission and Block II development will be closely monitored by AES.

Another source of propellant leakage is through the propellant tank relief valve. However, since the burst disc must be failed before leakage can occur, its analysis is part of reliability assessment. A third source is through the tank door seal. The leakage rates are specified as follows:

Engine valve ball leakage = 120 scc/hr (as nitrogen)

Engine valve shaft leakage = 40 scc/hr (as nitrogen)

Propellant tank leakage   = 820×10^{-6} scc/sec (as nitrogen)

This leakage was converted to an equivalent sonic orifice size using an upstream pressure of 240 psia. By assuming a discharge coefficient of 1.0 and an upstream pressure of 240 psi, the 45-day leakage of equivalent propellant was calculated to be less than 11 pounds.

Assuming that all sources leaked helium simultaneously for 45 days at the specification leak rate, less than 0.1 percent of the helium would be lost from the system on an AES mission. An increase by a factor of three to reflect three-sigma uncertainty would still not require an increase in helium loading.

The loss of N_2 due to leakage from each system during a 45-day coast period is approximately 0.027 pounds of N_2. This is equivalent to about three percent of the required N_2 load.

IN-FLIGHT LEAK DETECTION

Fluid leakage from the SPS during a 45-day mission is a problem that must be considered from the standpoint of both mission success and crew safety. For mission success, excess leakage must be prevented. For crew safety, excess leakage must be discovered at the earliest possible time so that the leak rate may be determined and abort action taken when necessary. Since the Block II outage control gauging system is inoperable during periods of coast, that is, during nonthrust or zero-gravity periods, crew safety improvement requires a separate leak detection system for SPS fluids.

Although the zero-gravity (coast) leak detection requirement has been established, the system for performing this function has not been determined. Many methods have been proposed, but it is not known at this time if any of the methods possess sufficient accuracy. For useful leak detection, the required accuracy is related to the quantity of contingency (reserve)

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propellant that may be aboard. If the gauge error exceeds the amount of reserve propellant (the excess over the amount required for abort) the gauge has no value. The required accuracy varies with mission flight plans, but, in general, is approximately two percent of the total propellant loaded and five percent of the total helium loaded. This desired accuracy appears to be feasible, but gauging capability will not be determined until the completion of the leak detection system feasibility studies.

Some of the characteristics of the SPS propellant system that must be considered to finally resolve the problem of leak detection accuracy are: thermal gradients, mass distribution gradients, random convection patterns for helium inflow, random liquid orientation and location during zero-gravity transfer line filling and draining for full and partially filled loadings, tank pressure fluctuations during firing due to regulator action and transfer line manometer effects, and helium inlet temperatures varying during blowdown to 20 F less than propellant temperature.

It is planned that the leak detection system will either supplant or utilize, as is, the existing outage control gauging system. Requirements applicable to the study are outlined below.

Leak detection gauging is defined as gauging during nonflow periods under conditions of zero to 5.0-g acceleration for durations up to 45 days. Changes in the fluid quantity (from that at the start of the nonflow period) contained in each of the two propellant systems (oxidizer and fuel) are to be measured and displayed. It is preferable that the system discriminate between propellant and helium quantity changes. It is also preferable that the system function continuously, but discrete measurements on command are adequate provided the operation is not limited to a fixed number of measurements. It is preferable but not mandatory that the measurement indicate total fluid quantities instead of merely changes, since the information could be used as a check against outage control gauging measurement obtained at the termination of the previous flow period.

Outage control gauging is defined as gauging during flow of helium into and propellant out of the propellant system under conditions of 0.1- to 1.0-g acceleration for durations of 1.0 to 630.0 seconds. Oxidizer and fuel remaining are measured and displayed separately and continuously during flow. The excess or deficiency of oxidizer remaining with respect to a 1.6 to 1 oxidizer-to-fuel mixture ratio is computed from these measurements and displayed continuously.

Mission requirements applying to the complete fluid gauging system include 45-day space flight operations which may be all coast (nonflow) period, or firings (flow) may be interspersed at any time in the 45-day

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period. Firings of less than 5 seconds will not require outage control gauging. Leak detection gauging is required from completion of servicing on the launch stack through boost and coast until the first SPS firing and for every coast period thereafter.

MATERIALS LIFE EXTENSION

To extend the Block II SPS capability from 14 to 45 days required review of SPS materials environmental exposure resistance. With one exception, the metallic materials used are considered to have inherent capability for 45 days since the basis for their selection was not significantly time limited. The exception is the use of titanium exposed to nitrogen tetroxide, a problem currently receiving intensive study by the Apollo program. The materials reviewed by AES were consequently the nonmetallics. A detailed review of the life capability of most of the nonmetallics is contained in NAA/S&ID report SID 65-1527. Since engine subcontractor participation was not included in this preliminary definition phase, only a partial list of representative engine nonmetallics were reviewed.

The study was limited to nonmetallic materials exposed to Aerozine 50 and nitrogen tetroxide propellants over the SPS design temperature range plus exposure to radiation as described by the NAA/S&ID report SID 65-1534, System Analysis Summary, and exposure to space vacuum.

The results of this study indicate, with a few exceptions, that the non-metallics of the Block II SPS are adequate for AES missions and environments. The exceptions are in the use of butyl rubber for the nitrogen tetroxide tank door seals (main door and sensor flange) and for the low temperature A-50 fuel/helium check valve. Since the material for the oxidizer/helium check valve has not been finally selected for Apollo, it has not yet been evaluated for AES. Radiation dosage expected for AES missions should have no appreciable effect since system materials are provided with the inherent shielding of component housings and spacecraft structure and equipment.

Propellant exposure demonstrations used for Apollo design verification and/or qualification tests have been limited to durations of 15 to 52 days, the period varying with the component. For this reason, supplementary tests of longer duration are recommended for AES component certification (see report SID 65-1148, General Test Plan).

The approach used for the study first required a listing of all non-metallic materials and compositions from Block II design drawings, specifications, and related data. Material compatibility with the propellants and radiation resistance data was then obtained from various test reports and published reports from industry. This data was then compared with the

extended mission requirement of exposure for 45 days and with radiation dosage occurring during the mission. Radiation dosage was determined using both a 1.0-percent and a 0.1-percent probability of occurrence of solar proton events and by calculating the effect of inherent shielding of the particular component. Other environment criteria, such as temperature limits, were considered to remain within the Block II specification requirements.

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AES SERVICE PROPULSION SYSTEM

Two configurations (designated A and B) of different propellant capacities are recommended for the AES SPS. Configuration A uses all four of the Apollo Block II propellant tanks. Configuration B uses only two of the tanks and is for off-loaded missions requiring less than 53 percent of the Block II propellant capacity. Configuration B is identical to Configuration A except for: (1) deletion of two propellant storage tanks, (2) plumbing and gauging changes necessitated by (1).

Both configurations are designed to provide the impulse necessary to change the linear momentum of the spacecraft for the normal and contingent AES mission maneuvers listed:

1. Abort after LES jettison (abort to orbit or abort to entry)

2. Earth orbital plane changes

3. Deorbit from earth orbit

4. Translunar course corrections

5. Translunar aborts

6. Lunar orbit insertion

7. Lunar orbital plane changes

8. Lunar orbit abort

9. Transearth injection

10. Transearth course corrections

11. Earth orbit injection

The SPS will provide the spacecraft velocity increments for these maneuvers by providing an approximately constant thrust in the plus X direction through the center of gravity of the spacecraft.

AES SPS

The AES baseline SPS is composed of the following integrated equipment as shown in Figure 57: engine, pressurization, propellant supply, propellant utilization and gauging system (PUGS), leak detection gauging, displays and controls, and electrical and instrumentation. The AES SPS configuration is identical to Block II except for changes discussed in a later section.

CONFIGURATION B

Configuration B of the AES SPS is the same as described for Configuration A with exceptions as noted in the following paragraph. A system fluid schematic is shown in Figure 58.

For Configuration B, the oxidizer is contained in a single Apollo Block II sump tank having an internal volume of 160.5 cubic feet and containing 12,948 pounds of usable oxidizer. Configuration of retention reservoir and lines downstream of the tank are identical to Configuration A. The fuel equipment is identical in size and configuration to the oxidizer equipment and contains 8092 pounds of usable fuel.

Because of the deletion of the two propellant storage tanks, Configuration B will require only two of the four tank sensors used on Configuration A. Some revision to the electronic and display portions of the PUGS as well as to the GSE may be required. System interfaces will remain unchanged, if possible, and changes to the PUGS/spacecraft/GSE will be minimized.

The use of only two sump tanks requires that the transfer line be eliminated and the one-inch helium supply line be adapted to the sump tank door propellant transfer line inlet. The transfer line drain and external disconnect coupling will remain for its orgiinal purpose but must be connected to the new helium supply line.

The GSE units used for check-out of the Configuration A SPS PUGS can be used for Configuration B check-outs with modifications necessitated by the removal of two propellant quantity gauging system probes.

DESIGN CHANGES FROM BLOCK II TO AES

The required changes in SPS design for both Configurations A and B from Apollo Block II to AES follow.

BANK A

BANK B

(E/13) SP8

INTERFACE

INTERFACE

FLEX SECTION

(E/25)

(E/25)

SP060IP (E/24)

(NC 2 WAY) (E/7)

GN₂

(E/24) SP0600P
PILOT PRE-VALVE
SP0666X

PILOT PRE-VALVE
SP0667X

GN₂

SP20

(E/11)

(E/23)

(E/23)

(E/26)

SPI (E/14)

(E/12)

SP22

(E/27) SPI2

(E/14) SPIO

SP26

(E/12)

PILOT VALVE #1
SP0662X

PILOT VALVE #2
SP0662X

SP24

(E/4)

PILOT VALVE #4
SP0664X

(E/8)

SP25

(E/12)

SP27

(NC 3 WAY) (E/7)

SP32

SP35

SP28

SP31

D

A

(E/7)

NO.1 SP33

SP34

NO.4

SP29

(E/6)

NO.2

NO.3

(E/3)

SP0060T

SP000X (E/29)

505 SP6

(E/17)

(E/19) SP066IP

501

INTERFACE

INTERFACE

SPI9
OXID

(E/2)

SP18
FUEL

SP004I

SP0040

SP0041

(E/9)

(E/9)

(E/10)

(E/10)

SP0009P (E/22)

(E/21) SP00IOP

(E/20) SP0020T

INTERFACE

SP5 (E/16)

SP0050T

SP0052T

SP005IT

SP0053T

(E/1)

506

ENGINE ASSEMBLY

E1. NOZZLE EXTENSION
2. THRUST CHAMBER ASSEMBLY
3. PROPELLANT VALVE-OXIDIZER (N.C.)
4. PROPELLANT VALVE-FUEL (N.C.)
5. PROPELLANT VALVE POSITION POT (REDUNDANT COIL)
6. PROPELLANT VALVE PNEUMATIC ACTUATORS
7. PROPELLANT CONTROL PILOT VALVE- SOLENOID ACTUATED N.C.
8. LOOP BALANCING ORIFICES
9. ENGINE SYSTEM BALANCING ORIFICES
10. PROPELLANT SCREEN
11. PRESSURE REGULATOR, PNEUMATIC (N_2)
12. FLOW CONTROL & TRIM ORIFICES
⑤13. PNEUMATIC FILL & VENT. GN_2 PRESSURIZING, SELF SEAL TYPE, DISCONNECT CAPPED FOR FLIGHT
14. REGULATOR OUTLET TEST PORT SELF SEAL TYPE DISCONNECT CAPPED FOR FLIGHT
15. PROPELLANT VALVE SEAL DRAIN (FUEL) OPEN TYPE QUICK DISCONNECT
16. PROPELLANT VALVE SEAL DRAIN. (OXIDIZER) OPEN TYPE, QUICK DISCONNECT
17. PROPELLANT INLET MANIFOLD FILL VENT. (OXID) SELF SEAL TYPE QUICK DISCONNECT WITH CAP FOR ENGINE OPERATION
18. PROPELLANT INLET MANIFOLD FILL VENT, (FUEL) SELF SEAL TYPE QUICK DISCONNECT WITH CAP FOR ENGINE OPERATION
19. THRUST CHAMBER PRESSURE BOSS
20. THRUST CHAMBER WALL TEMPERATURE SENSOR
21. FUEL INLET PRESSURE BOSS
22. OXIDIZER INLET PRESSURE BOSS
23. BURST DISC
24. PNEUMATIC STORAGE PRESSURE BOSS
25. PNEUMATIC STORAGE TANK (GN_2)
26. RELIEF VALVE
27. RELIEF VALVE TEST PORT SELF SEAL TYPE, DISCONNECT CAPPED FOR FLIGHT
28. GIMBAL ACTUATOR (YAW)
29. GIMBAL ACTUATOR (PITCH)

ENGINE SERVICE POINTS

SP 4- PROP. VALVE SEAL DRAIN (F.)
SP 5- PROP. VALVE SEAL DRAIN (O.)
SP 6- PROP. INLET MANIFOLD FILL
SP 7- PROP. INLET MANIFOLD FILL
SP 8- PNEUMATIC FILL & VENT GN
SP 9- PNEUMATIC FILL & VENT GN
SP10- REG. OUTLET TEST PORT
SP11- REGULATOR OUTLET TEST
SP12- RELIEF VALVE TEST PORT
SP13- RELIEF VALVE TEST PORT
SP18- FUEL INLET
SP19- OXIDIZER INLET
SP20- REG. AMBIENT SENSING PORT
SP21- REG. AMBIENT SENSING PORT
SP22- RELIEF VALVE OUTLET (A)
SP23- RELIEF VALVE OUTLET (B)
SP24- PILOT VALVE NO.1 VENT
SP25- PILOT VALVE NO.2 VENT
SP26- PILOT VALVE NO.3 VENT
SP27- PILOT VALVE NO.4 VENT
SP28- ACTUATOR NO.1 SPRING CAV.
SP29- ACTUATOR NO.2 SPRING CAV.
SP30- ACTUATOR NO.3 SPRING CAV.
SP31- ACTUATOR NO.4 SPRING CAV.
SP32- ACTUATOR NO.1 SHAFT SEAL
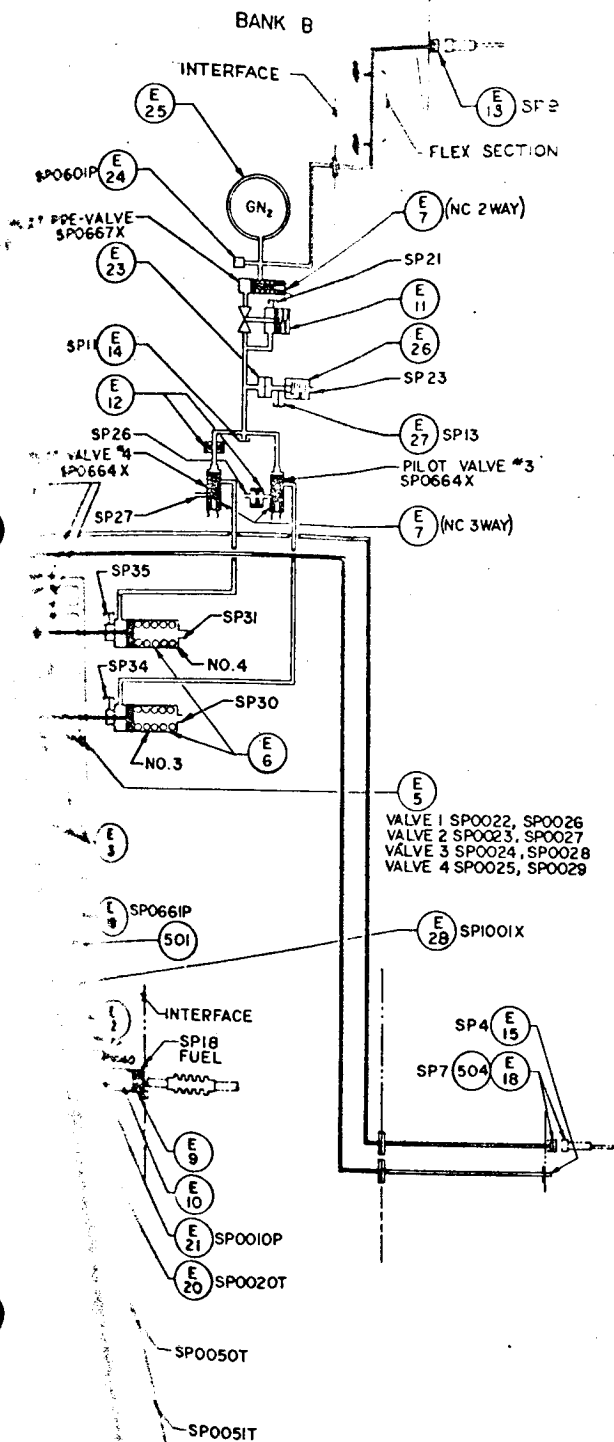SP33- ACTUATOR NO.2 SHAFT SEAL
SP34- ACTUATOR NO.3 SHAFT SEAL
SP35- ACTUATOR NO.4 SHAFT SEAL

BANK B

INTERFACE

FLEX SECTION

VALVE 1 SP0022, SP0026
VALVE 2 SP0023, SP0027
VALVE 3 SP0024, SP0028
VALVE 4 SP0025, SP0029

VALVE NUMBER	BALL DESIGNATION	DESCRIPTION	LOOP
1	B	LOWER UPSTREAM FUEL	
	F	LOWER DOWNSTREAM OX.	
2	D	LOWER DOWNSTREAM FUEL	PRIMARY
	H	LOWER UPSTREAM OX.	
3	C	UPPER DOWNSTREAM FUEL	
	G	UPPER UPSTREAM OX.	
4	A	UPPER UPSTREAM FUEL	SECONDARY
	E	UPPER DOWNSTREAM OX.	

SP	ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
SP 12&13		2	264004-4800	LEAR-SEIGLER	COUPLING,TEST REL.VALVE (GROUND)	
			(1)	AEROJET-GENERAL	COUPLING,TEST REL.VALVE (FLIGHT)	
SP 10&11		2	264004-4600	LEAR-SEIGLER	COUPLING, REG.OUTLET (GROUND)	
			(1)	AEROJET-GENERAL	COUPLING, REG.OUTLET (FLIGHT)	
SP 8&9	(5)	2	ME144-0023-0061	LEAR-SEIGLER	COUPLING, FILL VENT (GROUND)	MC144-0023
	(5)	2	ME144-0023-0071	LEAR-SEIGLER	COUPLING, FILL VENT (FLIGHT)	MC144-0023
	506	1	ME501-0132-0001	AEROJET-GENERAL	NOZZLE EXTENSION, SPS ENGINE	MC501-0009
SP6	505	1	ME273-0011-0002	J.C. CARTER	COUPLING FILL VENT & DRAIN (GROUND)	MC273-0011
		1	ME273-0011-0001		COUPLING FILL VENT & DRAIN (FLIGHT)	
SP7	504	1	ME273-0024-0002	J.C. CARTER	COUPLING FILL VENT (GROUND)	MC273-0011
		1	ME273-0024-0001		COUPLING FILL VENT (FLIGHT)	
	501	1	ME321-0004-000	AEROJET-GENERAL	ROCKET ENGINE-SERVICE PROPULSION	MC501-0009

ROCKET ENGINE SYSTEM

NTS

FUEL)
OXID.)
L VENT (OXID.)
LL VENT (FUEL)
GN₂) BANK A
GN₂) BANK B
(A)
PORT (B)
RT (A)
RT (B)

POPT (A)
PORT (B)
(A)
(B)
NT PORT
NT PORT
T PORT
T PORT
CAVITY VENT PORT
AVITY VENT PORT
AVITY VENT PORT
AVITY VENT PORT
EAL TEST PORT
EAL TEST PORT
EAL TEST PORT
SEAL TEST PORT

OXIDIZER SUMP #2

OXIDIZER 1 STORAGE

ENGINE ENV
SEE ZONE

TP17
TP12
TP14
TP25

SP0005

202 TP13

SP0003P
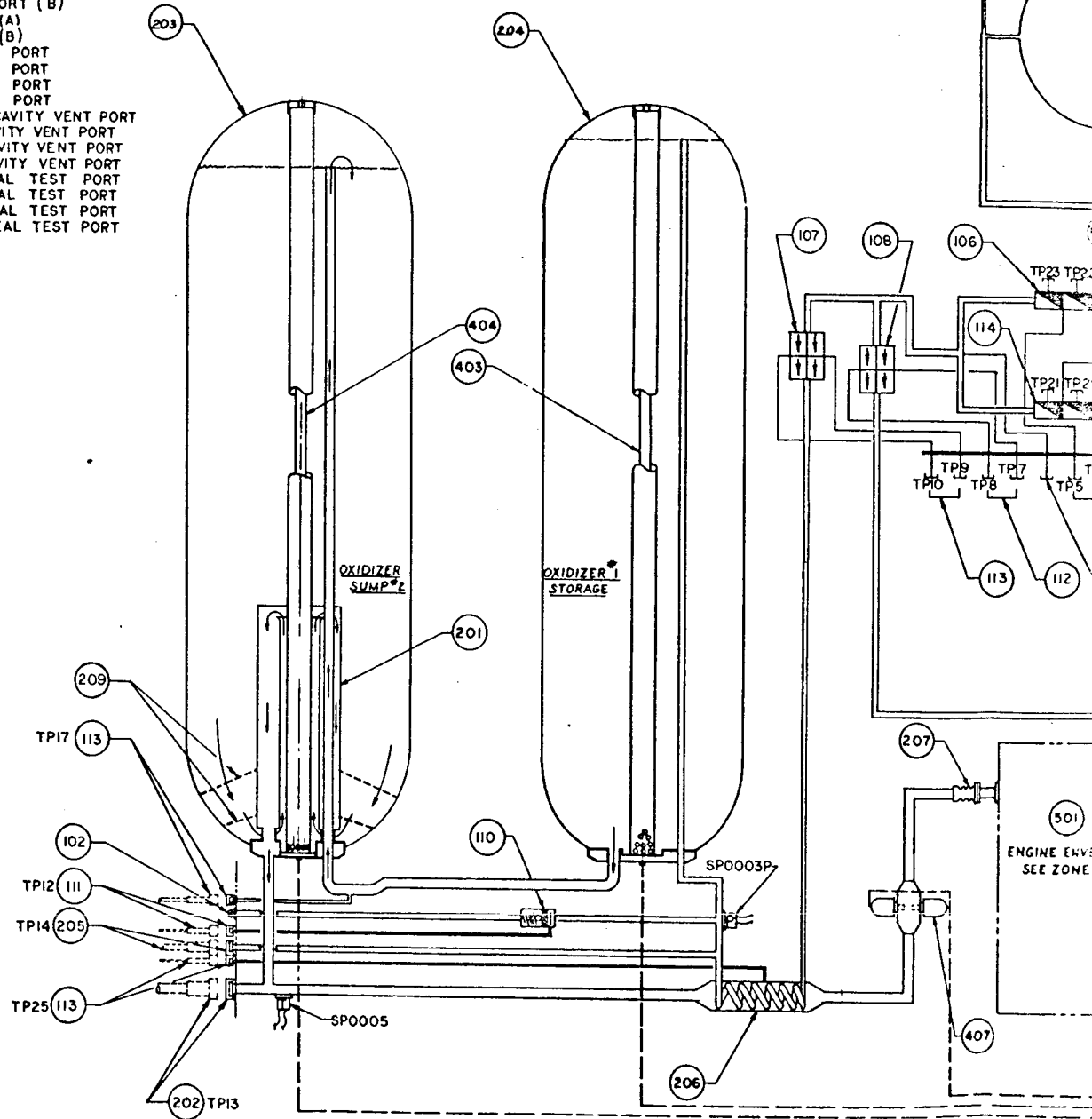
SP0655 Q QUANTITY SPS OXIDIZE
SP0656 Q QUANTITY SPS OXIDIZ
SP0657 Q QUANTITY SPS FUEL
SP0658 Q QUANTITY SPS FUEL

HELIUM

HELIUM

(103)

(401)

SP0011 Q TOTAL QUANTITY OXIDIZER
SP0012 Q TOTAL QUANTITY FUEL
SP0046 X PU VALVE INCREASE MONITOR
SP0047 X PU VALVE DECREASE MONITOR
SP0640 Q PROPELLANT UNBALANCE (OXIDIZER)
SP1002 X SPS PU SENSOR FAIL

(204)

(203)

(101-2)

SP-0001

TP22

SP-0031

(101-1)

TP20

SP0030

TP4 TP2

TP5 TP3

(104) TP1

(2) (111) TP6 (105)

(405)

(406)

FUEL #1
STORAGE

FUEL #2
SUMP

(201)

(209)

(306)

(112) TP

SP0006P

SP0006

(109)

(112) TP

(305)

(402)

(304) TP

5254-223

Figure 57. Se:

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
111	1 / 1	ME144-0023-006/ ME144-0023-0037	LEAR SIEGLER	COUPLING TP11 GROUND / COUPLING TP11 FLIGHT	MC144-0023
209	1	V17-470451	NAA	SCRN, PROPELLANT RETENTION	
201	1	V37-470240	NAA	RSVR, PROPELLANT RETENTION	
112	2 / 2	ME144-0023-0041 / ME144-0023-0031	LEAR SIEGLER	COUPLING, RESIDUAL DRAIN (GROUND) / COUPLING, RESIDUAL DRAIN (FLIGHT)	MC144-0023
306	1	ME273-0071-0001	AEROQUIP CORP	FLEX JOINT, FUEL LINE	MC273-0071
305	1	ME362-0019-0011	UNITED AIRCRAFT PROD	HEAT EXCHANGER, He LOW PRESS	MC362-0019
304	1 / 1	ME273-0020-0002 / ME273-0020-0001	J.C. CARTER	COUPLING, FILL & DRAIN (GROUND) / COUPLING, FILL & DRAIN (FLIGHT)	MC273-0012
203	1	V37-342102	NAA	TANK, FUEL SUMP	
204	1	V37-343102	NAA	TANK, FUEL STORAGE	
301	1 / 1	ME273-0012-0002 / ME273-0012-0001	J.C. CARTER	COUPLING, FUEL VENT (GROUND) / COUPLING, FUEL VENT (FLIGHT)	MC273-0012

FUEL DISTRIBUTION

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
111	1 / 1	ME144-0023-006/ ME144-0023-0037	LEAR SIEGLER	COUPLING TP12 GROUND / COUPLING TP12 FLIGHT	MC144-0023
209	1	V17-470451	NAA	SCRN, PROPELLANT RETENTION	
113	2 / 2	ME144-0023-0021 / ME144-0023-0011	LEAR-SIEGLER	COUPLING, OX RESIDUAL DRAIN (GROUND) / COUPLING, OX RESIDUAL DRAIN (FLIGHT)	MC144-0023
207	1	ME273-0040-0008	AEROQUIP CORP.	FLEX JOINT, OXIDIZER LINE	MC273-0040
206	1	ME362-0019-0001	UNITED AIRCRAFT PROD	HEAT EXCHANGER, He LOW PRESS	MC362-0019
205	1 / 1	ME273-0022-0002 / ME273-0022-0001	J.C. CARTER	COUPLING, OX VENT (GROUND) / COUPLING, OX VENT (FLIGHT)	MC273-0012
204	1	V37-343102	NAA	TANK, OXIDIZER STORAGE	
203	1	V37-342102	NAA	TANK, OXIDIZER SUMP	SID65-762
202	1 / 1	ME273-0018-0002 / ME273-0018-0001	J.C. CARTER	COUPLING, OX FILL & DRAIN (GROUND) / COUPLING, OX FILL & DRAIN (FLIGHT)	MC273-0012
201	1	V37-470240	NAA	RSVR, PROPELLANT RETENTION	

OXIDIZER DISTRIBUTION

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
114	1	ME284-0324-0001	B.H. HADLEY CO	REGULATOR, PRESS He (2 STAGE)	MC284-0324
113	2 / 2	ME144-0023-0021 / ME144-0023-0011	LEAR-SIEGLER	COUPLING, TP 9 & 10 (GROUND) / COUPLING, TP 9 & 10 (FLIGHT)	MC144-0023
112	2 / 2	ME144-0023-0041 / ME144-0023-0031	LEAR-SIEGLER	COUPLING, TP 7 & 8 (GROUND) / COUPLING, TP 7 & 8 (FLIGHT)	MC144-0023
111	1 / 1	ME144-0023-0061 / ME144-0023-0051	LEAR-SIEGLER	COUPLING, TP6 (GROUND) / COUPLING, TP6 (FLIGHT)	MC144-0023
110	1	ME284-0330-0001	CALMEC	VALVE, RELIEF OXIDIZER	MC284-0330
109	1	ME284-0330-0011	CALMEC	VALVE, RELIEF FUEL	MC284-0330
108	1	ME284-0327-0002	APCO	VALVE, QUAD CHECK, FUEL	MC284-0327
107	1	ME284-0327-0002	APCO	VALVE, QUAD CHECK, OXIDIZER	MC284-0327
106	1	ME284-0324-0001	B.H. HADLEY	REGULATOR, PRESS He (2 STAGE)	MC284-0324
105	4 / 4	ME144-0023-0091 / ME144-0023-0071	LEAR-SIEGLER	COUPLING, TP 2,3,4 & 5 (GROUND) / COUPLING, TP 2,3,4 & 5 (FLIGHT)	MC144-0023
104	1 / 1	ME273-0009-0002 / ME273-0009-0007	ON-MARK	COUPLING, FILL & DRAIN (GROUND) / COUPLING, FILL & DRAIN (FLIGHT)	MC273-0009
103	2	V37-347102	NAA	VESSEL, PRESSURE HELIUM	SID65-762
102	2	V17-460215	NAA	UNION, OVERBOARD RELIEF	
101	2	ME284-0318-0005	APCO	VALVE, SOLENOID SHUT-OFF	

PRESSURIZATION SYSTEM (HELIUM)

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
407	1	ME284-0219-0001	SIMMONDS PREC PROD	VALVE ASSEMBLY P.U.	MC901-0008
406	1	V37-320633-0001		SENSING PROBE FUEL TANK #2 (SUMP)	MC610-0001
405	1	V37-320632-0001		SENSING PROBE FUEL TANK #1 (STOR)	
404	1	V37-320631-0001		SENSING PROBE OX TANK #2 (SUMP)	
403	1	V37-320630-0001		SENSING PROBE OX TANK #1 (STOR)	
402	1	V37-320629-0001		CONTROL UNIT	
401	1	V37-320639-0001		DISPLAY PANEL C/M	MC610-0001

P.U. AND GAUGING SYSTEM

5. SUPPLIED BY NAA/SID AT INSTALLATION
4. REF. SID 65-1527 REPORT
3. REF. INSTL. DWGS V37-460201 HELIUM V37-470201 OXIDIZER
   V37-480201 (FUEL) V37-470202 (GAUGING) V37-410001 (ENGINE)
2. TP20, 21, 22 & 23 ARE ANDI0050-2 PORTS,
   PER SCD ME284-0324-0002
1. AIRBORNE HALF OF DISCONNECTS SUPPLIED BY AEROJET-
   GENERAL AS PART OF ENG. ASSY

NOTES: UNLESS OTHERWISE NOTED

TP18
111 TP11
102
301 TP16
TP24
TP15

5254-223
5254-223
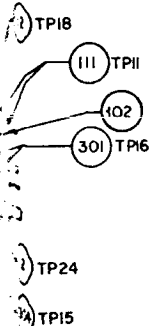
Service Propulsion Fluid Feed System for AES Phase II

ENGINE ASSEMBLY

E 1. NOZZLE EXTENSION
2. THRUST CHAMBER ASSEMBLY
3. PROPELLANT VALVE-OXIDIZER (N.C.)
4. PROPELLANT VALVE-FUEL (N.C.)
5. PROPELLANT VALVE POSITION POT (REDUNDANT COIL)
6. PROPELLANT VALVE PNEUMATIC ACTUATORS
7. PROPELLANT CONTROL PILOT VALVE-
    SOLENOID ACTUATED N.C.
8. LOOP BALANCING ORIFICES
9. ENGINE SYSTEM BALANCING ORIFICES
10. PROPELLANT SCREEN
11. PRESSURE REGULATOR, PNEUMATIC (N_2)
12. FLOW CONTROL & TRIM ORIFICES
(4)13. PNEUMATIC FILL & VENT. GN_2 PRESSURIZING, SELF SEAL
    TYPE, DISCONNECT CAPPED FOR FLIGHT
14. REGULATOR OUTLET TEST PORT SELF SEAL TYPE
    DISCONNECT CAPPED FOR FLIGHT
15. PROPELLANT VALVE SEAL DRAIN (FUEL).
    OPEN TYPE QUICK DISCONNECT
16. PROPELLANT VALVE SEAL DRAIN, (OXIDIZER).
    OPEN TYPE, QUICK DISCONNECT
17. PROPELLANT INLET MANIFOLD FILL VENT. (OXID)
    SELF SEAL TYPE QUICK DISCONNECT
    WITH CAP FOR ENGINE OPERATION
18. PROPELLANT INLET MANIFOLD FILL VENT, (FUEL)
    SELF SEAL TYPE QUICK DISCONNECT
    WITH CAP FOR ENGINE OPERATION
19. THRUST CHAMBER PRESSURE BOSS
20. THRUST CHAMBER WALL TEMPERATURE SENSOR
21. FUEL INLET PRESSURE BOSS
22. OXIDIZER INLET PRESSURE BOSS
23. BURST DISC
24. PNEUMATIC STORAGE PRESSURE BOSS
25. PNEUMATIC STORAGE TANK (GN_2)
26. RELIEF VALVE
27. RELIEF VALVE TEST PORT SELF SEAL TYPE,
    DISCONNECT CAPPED FOR FLIGHT
28. GIMBAL ACTUATOR (YAW)
29. GIMBAL ACTUATOR (PITCH)

ENGINE SERVICE P...

SP 4 - PROP VALVE SEAL DRAIN
SP 5 - PROP VALVE SEAL DRAIN
SP 6 - PROP INLET MANIFOLD FIL
SP 7 - PROP. INLET MANIFOLD F
SP 8 - PNEUMATIC FILL & VENT
SP 9 - PNEUMATIC FILL & VENT
SP10 - REG. OUTLET TEST POR
SP11 - REGULATOR OUTLET TES
SP12 - RELIEF VALVE TEST PO
SP13 - RELIEF VALVE TEST PO
SP18 - FUEL INLET
SP19 - OXIDIZER INLET
SP20 - REG. AMBIENT SENSING
SP21 - REG. AMBIENT SENSING
SP22 - RELIEF VALVE OUTLET
SP23 - RELIEF VALVE OUTLET
SP24 - PILOT VALVE NO.1 VEN
SP25 - PILOT VALVE NO.2 VEN
SP26 - PILOT VALVE NO.3 VEN
SP27 - PILOT VALVE NO.4 VEN
SP28 - ACTUATOR NO.1 SPRING
SP29 - ACTUATOR NO.2 SPRING
SP30 - ACTUATOR NO.3 SPRING
SP31 - ACTUATOR NO.4 SPRING
SP32 - ACTUATOR NO.1 SHAFT S
SP33 - ACTUATOR NO.2 SHAFT S
SP34 - ACTUATOR NO.3 SHAFT S
SP35 - ACTUATOR NO.4 SHAFT S

SP9
FLEX SECTION
(NC 2 WAY)
SP21
SP23
SP13
PILOT VALVE #3
SP0664X
(NC 3 WAY)
SP30
VALVE 1 SP0022, SP0026
VALVE 2 SP0023, SP0027
VALVE 3 SP0024, SP0028
VALVE 4 SP0025, SP0029
SP1001X
INTERFACE
SP18 FUEL
SP4
SP7 504
SP0010P
SP0020T
SP0050T
SP0051T
506

VALVE NUMBER	BALL DESIGNATION	DESCRIPTION	LOOP
1	B	LOWER UPSTREAM FUEL	PRIMARY
	F	LOWER DOWNSTREAM OX	
2	D	LOWER DOWNSTREAM FUEL	
	H	LOWER UPSTREAM OX	
3	C	UPPER DOWNSTREAM FUEL	SECONDARY
	G	UPPER UPSTREAM OX	
4	A	UPPER UPSTREAM FUEL	
	E	UPPER DOWNSTREAM OX	

SP			PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
SP 12 & 13		2	264004-4800	LEAR-SEIGLER	COUPLING,TEST,REL.VALVE (GROUND)	
			(1)	AEROJET-GENERAL	COUPLING,TEST,REL.VALVE (FLIGHT)	
SP 10 & 11		2	264004-4400	LEAR-SEIGLER	COUPLING, REG.OUTLET (GROUND)	
			(1)	AEROJET-GENERAL	COUPLING, REG.OUTLET (FLIGHT)	
SP8 & 9	(4)	2	ME144-0023-0081	LEAR-SEIGLER	COUPLING, FILL VENT (GROUND)	ME144-0023
	(4)	2	ME144-0023-0071	LEAR-SEIGLER	COUPLING, FILL VENT (FLIGHT)	ME144-0023
	506	1	ME501-0192-0001	AEROJET-GENERAL	NOZZLE EXTENSION, SPS ENGINE	MC501-0009
SP6	505	1	ME273-0011-0002	J.C. CARTER	COUPLING,ENG.VENT (DRAIN) (GROUND)	MC273-0011
		1	ME273-0011-0001		COUPLING,ENG.VENT & DRAIN (FLIGHT)	
SP7	504	1	ME273-0024-0002	J.C. CARTER	COUPLING, FILL VENT (GROUND)	MC273-0011
		1	ME273-0026-0001		COUPLING, FILL VENT (FLIGHT)	
	501	1	ME321-0004-0001	AEROJET-GENERAL	ROCKET ENGINE-SERVICE PROPULSION	MC501-0009
ITEM	QTY		PART NUMBER	SUPPLIER	DESCRIPTION	SPEC

ROCKET ENGINE SYSTEM

5254-214

6 5 4

HELIUM

HELIUM

(103)

(401)

SP0011 Q TOTAL QUANTITY OXIDIZER
SP0012 Q TOTAL QUANTITY FUEL
SP0046 X PU VALVE INCREASE MONITOR
SP0047 X PU VALVE DECREASE MONITOR
SP0640 Q PROPELLANT UNBALANCE (OXIDIZER)
SP1002 X SPS PU SENSOR FAIL

(203)

(406)

(101-2)
SP-0001
TP22
SP-0031
(101-1)
SP0030
TP14 TP2
TP5 TP3
(104) TP1

FUEL
SUMP

(201)

(209)

(?) (111) TP6 (105)

(306)

(112) TP18
(111) TP11
(102)
(301) TP16

SP0006P

(109)

SP0006

(112) TP24

(305)

(304) TP15

(402)

5254-21

5 4

Figure 58. Service Prop

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
111	1	ME144-0023-0061 / ME144-0023-0031	LEAR SIEGLER	COUPLING TP11 GROUND / COUPLING TP11 FLIGHT	MC144-0023
209	1	V17-470451	NAA	SCRU, PROPELLANT RETENTION	
201	1	V37-470240	NAA	RSVR, PROPELLANT RETENTION	
112	2	ME144-0023-0041 / ME144-0023-0031	LEAR SIEGLER	COUPLING, RESIDUAL DRAIN (GROUND) / COUPLING, RESIDUAL DRAIN (FLIGHT)	MC144-0023
306	1	ME273-0071-0001	AEROQUIP CORP	FLEX JOINT, FUEL LINE	MC273-0071
305	1	ME362-0019-0011	STD AIRCRAFT PROD	HEAT EXCHANGER, He LOW PRESS	MC362-0019
304	1	ME273-0020-0002 / ME273-0020-0001	J.C.CARTER	COUPLING, FILL & DRAIN (GROUND) / COUPLING, FILL & DRAIN (FLIGHT)	MC273-0012
203	1	V37-342102	NAA	TANK, FUEL SUMP	
301	1	ME273-0012-0002 / ME273-0012-0001	J.C.CARTER	COUPLING, FUEL VENT (GROUND) / COUPLING, FUEL VENT (FLIGHT)	MC273-0012

FUEL DISTRIBUTION

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
111	1	ME144-0023-0061 / ME144-0023-0031	LEAR SIEGLER	COUPLING TP12 GROUND / COUPLING TP12 FLIGHT	MC144-0023
209	1	V17-470451	NAA	SCRU, PROPELLANT RETENTION	
113	2	ME144-0023-0021 / ME144-0023-0011	LEAR-SIEGLER	COUPLING, OX RESIDUAL DRAIN (GROUND) / COUPLING, OX RESIDUAL DRAIN (FLIGHT)	MC144-0023
207	1	ME273-0040-0001	AEROQUIP CORP	FLEX JOINT, OXIDIZER LINE	MC273-0040
206	1	ME362-0019-0001	STD AIRCRAFT PROD	HEAT EXCHANGER, He LOW PRESS	MC362-0019
205	1	ME273-0022-0002 / ME273-0022-0001	J.C.CARTER	COUPLING, OX VENT (GROUND) / COUPLING, OX VENT (FLIGHT)	MC273-0012
203	1	V37-342102	NAA	TANK, OXIDIZER SUMP	SID65-762
202	1	ME273-0018-0002 / ME273-0018-0001	J.C.CARTER	COUPLING, OX FILL & DRAIN (GROUND) / COUPLING, OX FILL & DRAIN (FLIGHT)	MC273-0012
201	1	V37-470240	NAA	RSVR, PROPELLANT RETENTION	

OXIDIZER DISTRIBUTION

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
114	1	ME284-0324-0001	B H HADLEY CO	REGULATOR, PRESS He (2 STAGE)	MC284-0324
113	2	ME144-0023-0021 / ME144-0023-0011	LEAR-SIEGLER	COUPLING, TP 9 & 10 (GROUND) / COUPLING, TP 9 & 10 (FLIGHT)	MC144-0023
112	2	ME144-0023-0041 / ME144-0023-0031	LEAR-SIEGLER	COUPLING, TP 7 & 8 (GROUND) / COUPLING, TP 7 & 8 (FLIGHT)	MC144-0023
111	1	ME144-0023-0061 / ME144-0023-0051	LEAR-SIEGLER	COUPLING, TP 6 (GROUND) / COUPLING, TP 6 (FLIGHT)	MC144-0023
110	1	ME284-0330-0001	CALMEC	VALVE, RELIEF OXIDIZER	MC284-0330
109	1	ME284-0330-0011	CALMEC	VALVE, RELIEF FUEL	MC284-0330
108	1	ME284-0327-0002	APCO	VALVE, QUAD CHECK FUEL	MC284-0327
107	1	ME284-0327-0001	APCO	VALVE, QUAD CHECK OXIDIZER	MC284-0327
106	1	ME284-0324-0002	B.H. HADLEY	REGULATOR, PRESS He (2 STAGE)	MC284-0324
105	4	ME144-0023-0081 / ME144-0023-0071	LEAR-SIEGLER	COUPLING, TP 2,3,4 & 5 (GROUND) / COUPLING, TP 2,3,4 & 5 (FLIGHT)	MC144-0023
104	1	ME273-0009-0002 / ME273-0009-0001	ON-MARK	COUPLING, FILL & DRAIN (GROUND) / COUPLING, FILL & DRAIN (FLIGHT)	MC273-0009
103	2	V37-347102	NAA	VESSEL, PRESSURE HELIUM	SID65-762
102	2	V17-460215	NAA	UNION, OVERBOARD RELIEF	
101	2	ME284-0418-0805	APCO	VALVE, SOLENOID SHUT-OFF	

PRESSURIZATION SYSTEM (HELIUM)

ITEM	QTY	PART NUMBER	SUPPLIER	DESCRIPTION	SPEC
407	1	ME284-0219-0001	SIMMONDS PREC PROD	VALVE ASSEMBLY P.U.	MC901-0009
406	1	ME432-0033-0001		SENSING PROBE FUEL TANK #2 (SUMP)	MC610-0001
404	1	ME432-0031-0001		SENSING PROBE OX TANK #2 (SUMP)	
402	1	ME430-0008-0001		CONTROL UNIT (MODIFIED)	
401	1	ME434-0039-0001		DISPLAY PANEL C/M	MC610-0001

P.U. AND GAUGING SYSTEM

(4) 4. SUPPLIED BY NAA/SID AT INSTALLATION
   3. REF. SID65-1527 REPORT
   2. TP20,21,22 & 23 ARE ANDIO050-2 PORTS, .
      PER SCD ME284-0324-0002
(1) 1. AIRBORNE HALF OF DISCONNECTS SUPPLIED BY AEROJET-
      GENERAL AS PART OF ENG. ASSY ME321-0004-0001

NOTES: UNLESS OTHERWISE NOTED

5254-214

5254-214

Service Propulsion Fluid Feed System (Two Tanks)

SID 65-1519

Changes will be made to the engine and its installation to allow for external servicing of its pneumatic system.

Due to space allocations on the AES SM, the helium panel located in Sector IV of Block II must be rearranged. However, the relationship of the mainstream components and lines will remain unchanged. Panel structure and test point configuration will be changed but will not require retest.

A feasibility study is planned covering the design of a new or supplementary fluid (propellant and helium) gauging system to allow continuous quantity readout capability for leak detection purposes. The present gauging system function of providing outage control information will be retained and performed by the new system or by the present Block II system depending on the outcome of the study.

Analysis has indicated that backflow propellant leakage through the helium check valve will be a problem during a 45-day AES mission even through such leakage can be tolerated during the Apollo Block II missions. System changes to eliminate this problem are required. Future studies will determine the extent and nature of such changes.

POTENTIAL AES CHANGES FROM BLOCK II

Apollo test results for the present butyl rubber propellant tank door seal appear to have acceptable leak rates (maximum 5.5×10^{-4} scc/sec), but the effect of N_2O_4 on the O-ring (shore, volume swell, and elongation) are of such magnitude as to cause doubt for any usage beyond the 16 days of the Apollo test.

Implementation of a new door seal design is presently being accomplished for the Apollo propellant tanks. Final resolution in this area will be obtained after Apollo door seal testing and design revision has been accomplished.

The propellant valve shaft seat drain line may require some modification because of propellant leakage with possible freezing and blockage of the line. The propellant trapped in the blocked line can leak into the common gear housing, causing a possible corrosive or explosive condition or impair the integrity of the valve shaft seal.

After final selection of helium check valve poppet seal material for Block II, the material must be reevaluated for AES 45-day propellant exposure resistance.

SUMMARY

The AES service propulsion subsystem is essentially the same as the Apollo Block II SPS. The changes required are:

1. Addition of a zero-gravity leak detection helium and propellant gauging system to improve crew safety (Block II gauging system is inoperable during periods of coast.)

2. Provision for external nitrogen servicing for the engine to permit more efficient space utilization in the SM.

3. Change in the helium pressurization system to eliminate backflow leakage of propellant to the regulator.

4. Use of an alternate, reduced-propellant-tankage configuration to save weight on off-loaded flights. (This configuration eliminates the propellant storage tanks and uses only the sump tanks.)

5. Minor rearrangement of nonflow components on the helium panel to permit changes in the panel configuration required by AES SM space allocations.

In addition, potential changes for AES exist in areas still under design revision and development for Apollo Block II, depending on the design selected by Apollo. These include the oxidizer tank door seal (a change in material is required), the engine valve shaft seal drain (a heater may be required), and the oxidizer and fuel check valves (a change in material may be required).

These changes are the result of preliminary evaluation of AES requirements. AES missions have been designed to fall within the maximum propulsion capability of the Block II SPS, and no changes in SPS primary performance characteristics are required. However, the AES mission duration of 45 days, when compared to the Apollo Block II duration of 14 days, requires review of materials life, leakage effects, and coast-dependent failure effects on reliability. In addition, the AES SM incorporates minor changes from the Apollo Block II SM, and the SPS installation requirements required review.

SID 65-1519

Materials life was evaluated for 45-day exposure to propellants, radiation, and high vacuum. None of these environments, when extended from 14 to 45 days, changes the choice of Block II SPS materials except in areas where new materials are being selected for Block II.

The AES meteoroid environment requires protection of propellant tanks and lines and an investigation of engine nozzle extension failure modes with meteoroid penetration.

The increase in mission duration has several implications on SPS fluid leakage. Crew safety will be improved by the addition of a zero-gravity leak-detection system to provide information for timely abort action. For improvement of mission success, only leak prevention measures are useful. This area requires further study of design aspects of seals and seal redundancy, plus improvements in acceptance test technique for predicting production article leak rates over long missions. The Block II leak rate standards for acceptance test were reviewed and found to be adequate except for backflow leakage to the regulator.

For AES missions requiring less than a full propellant load, a reduced propellant tankage trade study was conducted. This resulted in a configuration (designated as SPS baseline Configuration B) to be used as an alternate to the standard Block II propellant tank configuration which is required for some AES missions. The final selection, consisting of two unmodified Block II sump tanks, was chosen primarily for its simplicity.

Other configurations of less capacity would cover the presently anticipated off-loaded mission propellant requirements and with greater weight savings. However, the increased complexity of their changes required much greater cost per pound of weight saved. The recommended configuration is applicable to those AES missions requiring less than 21,000 pounds of SPS propellant. It uses only the two sump tanks, the two SPS propellant storage tanks being omitted. Both of the Block II helium vessels are retained. However, it is recommended that the definition of Configuration B be changed to use only one Block II helium vessel. This will result in a 10-percent decay in chamber pressure and thrust over the last 30 seconds of flights requiring 21,000 pounds of propellant.

The recommended configuration has a capacity for 53 percent of the maximum usable propellant load, saves 586 pounds (the saving is 864 pounds if the change to one helium vessel is made), and has the lowest cost per pound of burned weight saved.

REACTION CONTROL SYSTEM

REACTION CONTROL SYSTEM

The objectives of the AES reaction control system studies were to evaluate the Apollo status of the command module and service module RCS subsystems, evaluate and relate the AES mission requirements and environments to the Apollo subsystem designs, determine required subsystem changes to meet AES mission demands, and identify those design and analysis areas of the subsystems that would require more extensive investigation.

Studies of both the command module and service module reaction control subsystems (RCS) were conducted.

The Apollo command module and service module RCS subsystems including capabilities and status are discussed herein. The space and propellant exposure capabilities of the subsystem and its components are presented with respect to the AES mission environments. Following discussions on the propellant and engine requirements of the AES service module RCS subsystem, trade-off studies are described, wherein a service module RCS configuration for AES is recommended. Using the recommended service module RCS configuration, the requirements and alternatives are outlined for nonnuclear gaging of the propellant quantity.

The requirements for modifications had to be based on AES housekeeping and transit functions and not experimental requirements. Both housekeeping and experimental requirements in terms of propellant, burn time, and engine starts are shown in this section; however, only housekeeping requirements are used to establish system modifications.

The following section presents a summary of the AES RCS subsystem studies conducted during the preliminary definition phase. A more detailed description of the studies concerning the RCS subsystems is presented in SID 65-1528, the reaction control system document of this report.

AES SM RCS PROPELLANT REQUIREMENTS

Preliminary service module RCS propellant requirements for the four AES reference missions are outlined and discussed in SID 65-1520, the guidance and control system document of this report. The work discussed here considers vehicle configuration, moments of inertia, and service module RCS moment arms; it compares these vechicle considerations with the mission maneuver requirements and engine performance. The specific

ground rules on which the propellant requirements are based include consideration of G&N and SCS duty cycles; attitude dead bands and maneuver rates; sources of maneuver simulation data; roll rates for vehicle thermal control; and maneuvers required by experiments.

Propellant requirements established in SID 65-1520 and used herein assumed the following values for engine specific impulse:

1. An I_{sp} of 279 seconds for all translational maneuvers.

2. Automatic rotational maneuvers are calculated, assuming an average I_{sp} of 238 seconds.

3. Manual rotational maneuvers are taken from Apollo simulation data. Analysis of one such simulation leads to an average I_{sp} of 238 for such maneuvers.

4. Attitude hold to be accomplished with single jet firings in both the G&N and SCS modes. The minimum impulse bit from a single jet is assumed to be 1.2 lbf-sec, which is about the upper limit of current engine test experience. An I_{sp} of 170 seconds was used in attitude hold calculations.

The propellant requirements discussed herein include propellant for contingencies selected in a manner consistent with current Apollo Block II practice. A 10-percent reserve has also been included in all propellant quantities to account for (1) jet impingement; (2) SPS propellant slosh; (3) torques from venting gases; (4) control system configuration variation; (5) disturbances from internal rotating machinery, and (6) RCS engine performance variation caused by propellant temperature changes, firing voltage, engine-to-engine differences, and other effects.

REFERENCE MISSION PROPELLANT REQUIREMENTS

Propellant requirements for each of the four reference missions are summarized below. Total propellant quantity is presented for each of the following conditions:

1. Propellant required for the translunar and transearth phases of lunar missions (ascent and descent phases of earth orbital missions) plus housekeeping requirements only for orbital phases. Contingency propellant is included for the return phase only.

2. Propellant for item 1 plus estimated requirements for experiments, assuming no service module RCS quad failure.

3. Propellant for item 2 plus contingency propellant to permit failure of one SM RCS quad at the beginning of the mission without reduction in mission capability.

The total propellant requirements are shown in Table 54 for normal housekeeping functions and housekeeping and experimental functions for the four reference missions.

Table 54.　Total Propellant Requirements

AES Mission	Housekeeping Functions Only		Housekeeping and Experimental Functions	
	Total Propellant Requirements	O/F Ratio	Total Propellant Requirements	O/F Ratio
1	1703.5	1.88	2124.8	1.82
2	1011.3	1.89	1841.9	1.87
3	869.0	1.88	1993.7	1.85
4	1101.9	1.76	NA	NA

From the preceding data, the average oxidizer-to-fuel ratio (O/F) for the first three missions is seen to be 1.88 for housekeeping functions and 1.85 when experiments are added.

As previously estimated, the four reference missions require propellant totals of 1704, 1011, 869, and 1102 lbm for housekeeping functions. When estimated experimental requirements are added to the first three missions, these totals become 2125, 1842, and 1994 lbm. If the contingency of one-quad-out at liftoff without mission degradation is allowed for the first three missions, these totals become 2886, 2533, and 2665 lbm. In each case, the requirement for reference mission 1 is the greatest; however, it is possible to reduce the total required by mission 1 by at least 100 lbm (as described in detail in SID 65-1528).

It becomes apparent that at least 1600 lbm propellant must be provided to the service module RCS for housekeeping functions only and that provision of 2025 lbm will permit the service module RCS to perform all estimated experimental requirements for the four reference missions as well. Accomplishing both housekeeping and experimental functions with early loss of a service module RCS quad could require up to 2800 lbm of propellant.

ENGINE STARTS AND BURN TIMES

The number of engine starts per service module RCS jet and the burn time per jet have been calculated for the four AES reference missions. These engine starts are based on the same ground rules used to determine

mission requirements in the preceding section. The 10-percent increase is considered just as appropriate for burn time as for propellant requirements; it is also not unreasonable for engine starts if considering the uncertainty in the number of pulses per maneuvers.

Engine starts and burn time have been calculated for three conditions for each of the reference missions:

1. Requirements for the translunar and transearch phases of lunar missions (ascent and descent phases of earth orbital missions) plus housekeeping requirements only for orbital phases. Contingency requirements are included for the return phase only.

2. Requirements for item 1 plus estimated requirements for experiments, assuming no service module RCS quad failure.

3. Requirements for item 2 plus contingency requirements to permit failure of one service module RCS quad at the beginning of the mission without reduction in mission capability.

Start and burn time estimates are reported separately for pitch, yaw, and roll jets. The roll jet estimates assume only four engines are used for roll control; the total number of roll starts, therefore, may be halved on the assumption that roll will be shared equally among eight jets.

Where requirements for plus and minus jets are different, the worst case has been listed. This applies only to the pitch and yaw jets because of differences between missions in estimated requirements for +X and -X translation. The difference in total jet burn time between +X and -X jets is not major, and the effect on engine starts is insignificant.

AES mission duty cycle information is not yet available for determining operating profiles in terms of on-time and off-time.

Engine Starts

Worst-case engine start requirements per jet are summarized in Table 55, with the number of the reference mission in which the worst case occurs in parentheses following the number of starts.

Burn Time Per Jet

Worst-case engine burn time requirements per jet are summarized in Table 56, with the number of the reference mission in which this worst case occurs shown in parentheses.

Table 55. Engine Starts per Jet

Mission Function	Roll (4 jets)	Roll (8 jets)	Pitch (4 jets)	Yaw (4 jets)
Housekeeping	4373(3)	2187(3)	2331(3)	3811(3)
Housekeeping and experiments	10625(1)	5313(1)	9878(3)	8893(3)
Housekeeping and experiments with quad out from liftoff	14136(1)	7068(1)	13026(3)	11646(3)

Table 56. Burn Time per Jet (second)

Mission Function	Roll (4 jets)	Roll (8 jets)	Pitch (4 jets)	Yaw (4 jets)
Housekeeping	217(2)	109(2)	510(1)	496(1)
Housekeeping and experiments	362(2)	181(2)	739(3)	568(1)
Housekeeping and experiments with quad out from liftoff	503(2)	127(2)	1010(3)	807(1)

SID 65-1519

SM RCS CONFIGURATION STUDIES

Service module RCS propellant requirements for the AES housekeeping functions are generally in excess of the Apollo Block II requirements and capacity. The propellant estimates are summarized in Table 57. For mission 1, the propellant was reduced from 1704 lbm to 1604 lbm by combining maneuvers as explained in SID 65-1528. To accommodate the increased quantity of propellant, the AES service module RCS configuration must be larger than the Apollo Block II. In this section, the recommended AES service module RCS configuration is developed by consideration of the following:

1. The optimum propellant tankage capacity and arrangement to meet previously discussed propellant requirements

2. The necessary helium supply to support the propellant tankage

3. A pressurization system compatible with the propellant and helium supply

4. The configuration and location of the engine clusters

PROPELLANT TANKAGE

New propellant tank designs or some combination of existing tanks could be used for the service module RCS. With newly designed tanks, it would be possible to consider a central service module location with a single

Table 57. Housekeeping Propellant Requirements

Mission Function	Propellant Quantity (lbm)			
	Mission 1	Mission 2	Mission 3	Mission 4
Housekeeping	1604	1011	869	1102
Housekeeping and experiments	2025	1842	1994	-
Housekeeping and experiments plus quad out from liftoff	2753	2533	2665	-

tankage and pressurization system feeding all four engine clusters. Such a central system could provide weight advantages and less stringent propellant budgeting (equal usage of propellant by each quad would not be needed). However, a central tankage concept conflicts with a "minimum-change" philosophy, competes with other systems for service module volume in sectors I, VI, or VII, requires lengthy propellant plumbing, and does not have the reliability advantages of the four-independent-quad concept.

Existing tanks that were considered for the AES configuration are the Apollo service module RCS and LEM RCS oxidizer and fuel tanks. Although there is some question concerning the capability of these bladder-expulsion type tanks for long mission durations, for the purpose of the configuration study it is assumed that the tanks will be qualified for the Apollo program and usable for AES. Therefore, use of Apollo and/or LEM propellant tanks in four independent service module RCS quads was assumed as a ground rule for the tank trade-off study. Since the LEM and Apollo RCS tanks are identical, except for length and external fill/vent details, and since they all use Teflon seals, it is appropriate to consider use of oxidizer tanks to carry fuel and fuel tanks to carry oxidizer.

Calculated LEM propellant tank capacities for use in the AES service module RCS and capacities of Apollo service module fuel tanks for oxidizer and oxidizer tanks for fuel were calculated and are summarized in Table 58.

The total available propellant for these tank capacities is shown in Table 59.

Candidate Tank Combinations

Estimated propellant requirements show that the propellant usage for mission 1 is greater than for the other three missions. Using mission 1, therefore, as the design basis, propellant tankage combinations that would be adequate for this mission were evaluated for the housekeeping functions only.

Mission 1 requires 1603.5 lbm of RCS propellant to support ascent, orbital housekeeping, and descent (the latter with two RCS quads out). This requires 400.9 lbm of propellant per quad, consisting of 139.1 lbm of fuel and 261.8 lbm of oxidizer. The minimum tankage to meet this requirement is:

Fuel: Two Apollo service module fuel tanks (139.4 lbm available propellant)

Oxidizer: Two Apollo service module oxidizer tanks (264.4 lbm available propellant)

Table 58.   Capacities of Apollo SM Tanks

RCS Tank	Fuel Capacity (lbm)	Oxidizer Capacity (lbm)
Apollo service module fuel	69. 7	108. 3
Apollo service module oxidizer	88. 7	138. 1
LEM fuel	101. 0	157. 4
LEM oxidizer	126. 2	196. 6

Table 59.   Total Available Propellants

RCS Tank	Available Fuel	Available Oxidizer
Apollo service module fuel	66. 9	103. 6
Apollo service module oxidizer	85. 2	132. 2
LEM fuel	97. 1	150. 8
LEM oxidizer	121. 4	188. 4

To base the design on the current estimate of housekeeping functions would seriously restrict vehicle capability.   The housekeeping functions are based on preliminary mission profiles that could be in error by as much as 30 percent.   Therefore, 30-percent additional propellant is provided to account for such a contingency.

Adding the 30 percent to the housekeeping function increases the required RCS propellant per quad to 506.2 lbm (179.5 lbm fuel and 326.7 lbm oxidizer).   The alternatives shown in Table 60 are available to contain this amount of propellant.

Alternative 1 appears undesirable from the standpoint of design complexity, gaging, and the logistics created by three tank sizes.   Alternative 5 was recommended at the time of the midterm briefing because its O/F ratio (1.6) was more compatible with the propellant requirements predicted at that time.   Alternatives 2, 3 and 4 appear to be equally capable

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Table 60. Alternatives to Contain Propellant

Alternative	Fuel Tankage	Oxidizer Tankage	Reserve Propellant (O/F=1.82)	Excess Propellant
1	182.3 lbm in one Apollo SM oxidizer and one LEM fuel tank	339.2 lbm in one LEM fuel and one LEM oxidizer tank	7.6 lbm	7.5 lbm oxidizer
2	194.2 lbm in two LEM fuel tanks	As in alternative 1	19.2 lbm	7.8 lbm fuel
3	As in alternative 2	376.8 lbm in two LEM oxidizer tanks	41.2 lbm	26.6 lbm oxidizer
4	218.5 lbm in one LEM fuel and one LEM oxidizer tank	As in alternative 3	66.2 lbm	11.2 lbm fuel
5	242.8 lbm in two LEM tanks oxidizer	As in alternative 3	66.2 lbm	35.5 lbm fuel

in the above trade-off, with alternative 3 (fuel in two LEM fuel tanks, oxidizer in two LEM oxidizer tanks) having the capability of being split into equal pairs.

Candidate Tank Arrangements

The obvious locations for service module RCS propellant tanks are the outside corners of service module sectors II, III, V, and VI as shown in Figure 59. Further, the 12.64-inch maximum outside diameter of Apollo and LEM tanks is tailored for these corner locations. Within these locations, four possible arrangements (T-1 through T-4) of the tanks were considered:

Arrangement T-1: Apollo Block II concept. One oxidizer, one fuel, and one helium tank in a single stack.

Arrangement T-2: Four propellant tanks (two on each side) attached to a full-sector door which is slightly more than half the length of the service module; helium tanks placed at the top of the door above the dome of the SPS tanks.
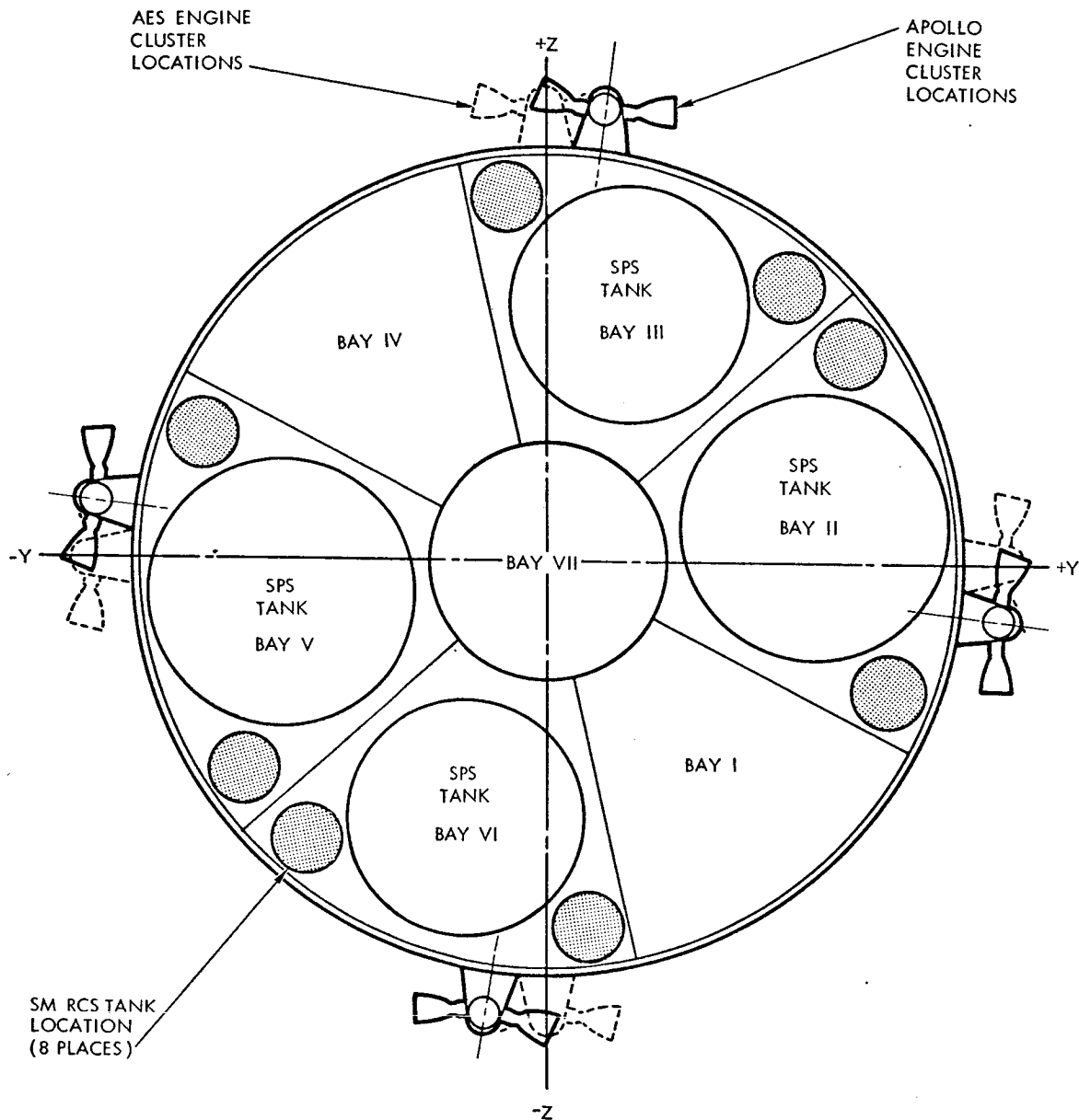
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Figure 59.　Potential SM-RCS Propellant Tank and Engine Quad Locations

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Arrangement T-3: Four propellant tanks in a single stack, extending the length of the service module; helium tanks cantilevered at top of stack.

Arrangement T-4: Two stacks of propellant tanks extended the full length of the service module; helium tanks located at top of sector door.

Each of the preceding arrangements is shown in Figure 60.

Tank Arrangement Trade-off

Reviewing the capabilities and design characteristics of the candidate tank arrangements, either arrangement T-2 or T-3 is capable of satisfying all housekeeping functions and the 30-percent factor of the AES missions if a service module RCS quad failure (the first) does not occur before some point late in the orbital phase; only arrangement T-4 is acceptable if all mission requirements must be met in spite of an early quad failure. The assumption was made that mission plans permit the CSM to satisfy house-keeping functions first and allot excess capacity to experimental functions. With this assumption, the trade-off between arrangements T-2, T-3, and T-4 reduces to the following considerations:

1. ECS Radiator Considerations: Only arrangement T-2 requires no change in ECS radiator design; arrangement T-4 would require a hardline connection when the service module RCS door is installed. Therefore, arrangement T-4 was rejected.

2. Propellant Tank Design Considerations: Arrangement T-3 would require modification of the LEM RCS tank flanges. Since some modification to the flanges is desirable for more efficient line routing (even in arrangement T-2), and since internal changes to the tank design are not involved, this is not a controlling consideration.

3. Structure and Weight Considerations: Arrangement T-3 carries with it a substantial weight penalty because of the structural weight needed to carry loads from the narrow cantilevered RCS door across to the service module skin at a point in the center of the sector. These structural considerations are described in more detail in SID 65-1529, the spacecraft design document of this report.

From the preceding considerations, it is apparent that the tank arrangement trade-off falls primarily in the design integration area, and is
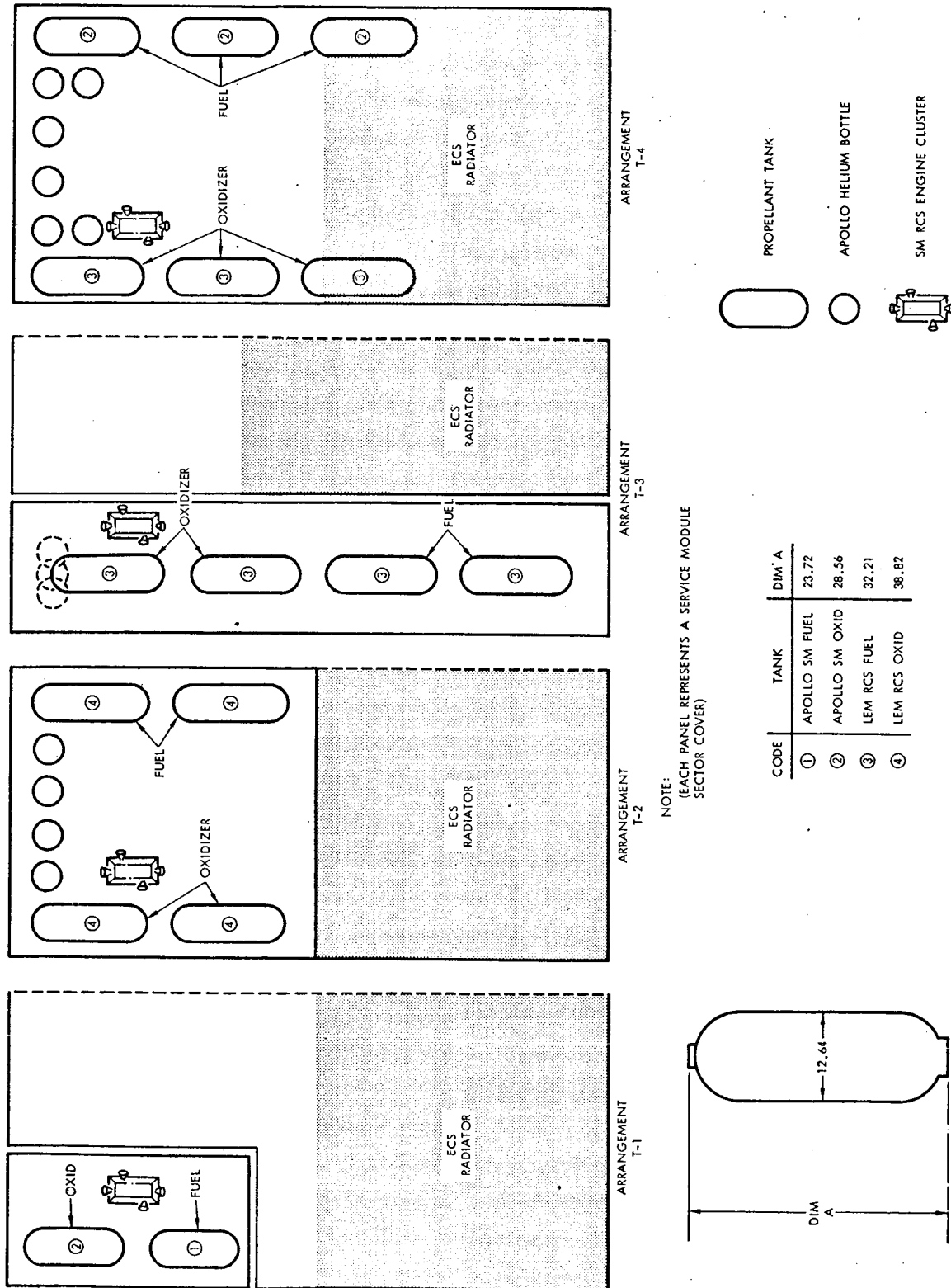
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Figure 60. Candidate SM-RCS Tank Arrangements

NOTE:
(EACH PANEL REPRESENTS A SERVICE MODULE SECTOR COVER)

CODE	TANK	DIM A
①	APOLLO SM FUEL	23.72
②	APOLLO SM OXID	28.56
③	LEM RCS FUEL	32.21
④	LEM RCS OXID	38.82

PROPELLANT TANK

APOLLO HELIUM BOTTLE

SM RCS ENGINE CLUSTER

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reported in SID 65-1529. The conclusion described therein, based primarily on weight and structural considerations, is that arrangement T-2 is clearly superior and is the recommended configuration for AES.

HELIUM REQUIREMENTS AND TANKAGE

With a service module RCS quad configuration containing four LEM oxidizer-size propellant tanks, it is necessary to determine the quantity of helium gas required to pressurize and expel the propellant in the tanks during the mission. It is also desirable to compare the capacity of existing Apollo and LEM helium tanks with the required quantity of helium in order to select the most favorable size and number of tanks for use with the AES configuration.

Helium Tankage Recommendation

The helium requirements based on the recommended configuration is 1.2796 lbm. Four LEM oxidizer-size propellant tanks require approximately 1.280 lbm of helium for pressurization. Three Apollo-size helium tanks contain a total of 1.553 lbm of available helium; this combination meets the requirements (with a reserve of 21.3 percent), and is used in the pressurization system trade-offs, except where redundant helium tankage is used in the system.

In the P-3 and P-4 pressurization system configurations, the helium system is divided into two sides; either side should be capable of supplying sufficient helium to the four LEM oxidizer-size tanks if the other side fails. Since one LEM helium tank, weighing 11.5 lbm maximum empty, contains only 0.902 lbm of available helium, and two Apollo helium tanks, weighing 10.5 lbm maximum empty, contain 1.035 lbm of available helium, the Apollo tanks are recommended for these configurations.

SM RCS Pressurization Subsystem Trade-offs

Five candidate configurations (P-2 through P-6) were selected for analysis and comparison with the Apollo Block II subsystem design (P-1). The configurations, compared schematically in Figure 61, are identified as follows:

Configuration P-1: Apollo Block II

Configuration P-2: The minimum possible modification to Apollo Block II

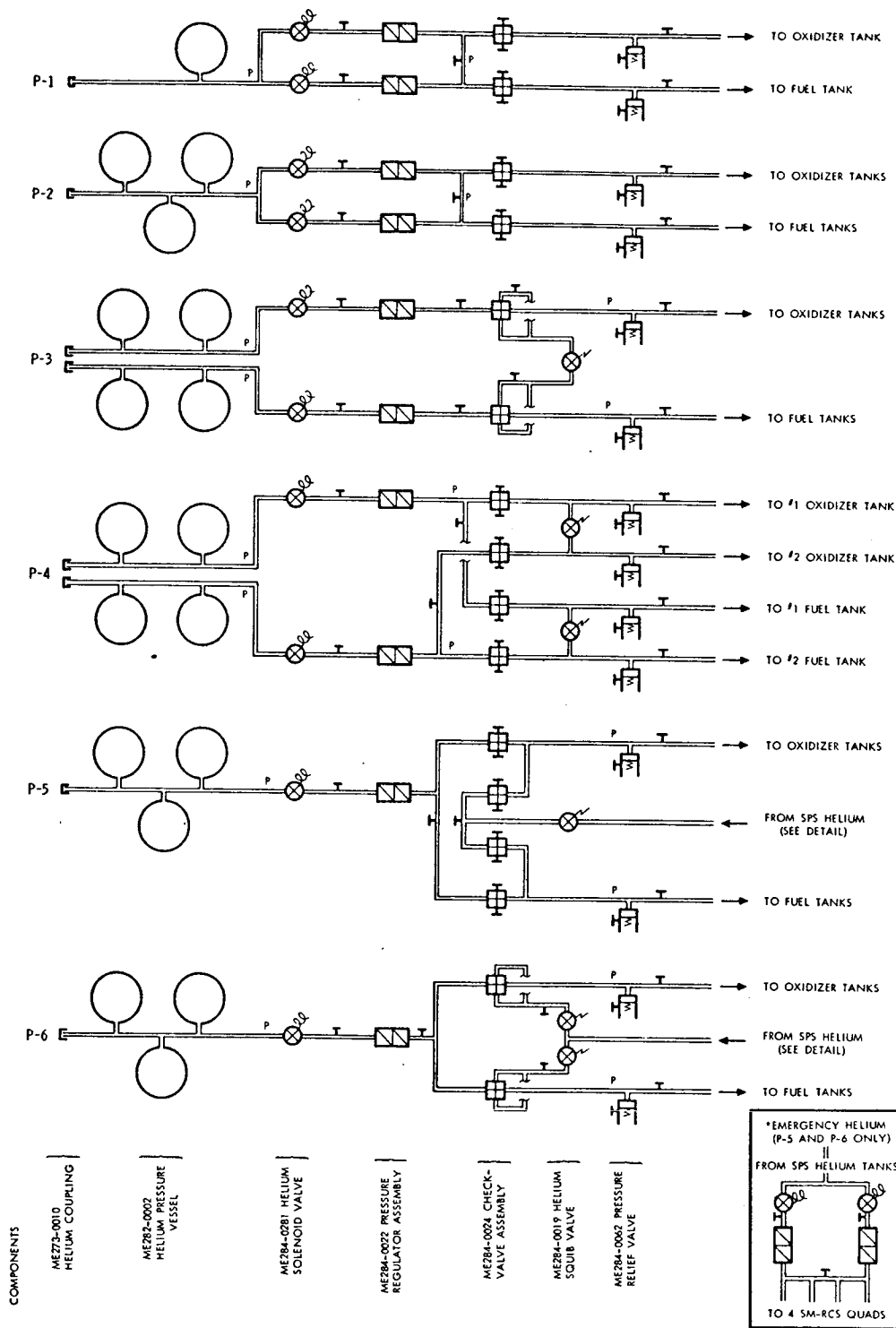Configuration P-3: In-quad helium redundancy by check valve interconnect

Figure 61.   SM-RCS Pressurization Configurations Studied

NOTES: "T" SIGNIFIES ME144-0023 TEST POINT COUPLING
        "P" SIGNIFIES PRESSURE TRANSDUCER

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Configuration P-4:　In-quad maximum reliability, using end-to-end redundancy

Configuration P-5:　Four-check-valve system, using service propulsion system helium to provide helium redundancy

Configuration P-6:　A three-check-valve version of P-5

The weight, reliability (from component apportionments), and technical advantages and disadvantages of the various configurations form the basis for the selection of configuration P-3 for AES.

CIRCUMFERENTIAL CLUSTER LOCATION

The service module RCS engine clusters in both Block I and II are located 7 deg, 15 min from the service module Y and Z axes, to make possible the location of the quads on an equal arc (20 degrees) from the sector I/II, III/IV, IV/V, and VI/I radial interfaces and, thereby, use a single RCS module design for all four positions (two of which are turned upside down relative to the other two). This 7 deg, 15 min (θ) rotation results in significant propellant penalties. When a pitch (yaw) force along the Y and Z is required, it acts over a moment arm of cos θ (0.992) times the service module radius. At the same time, an unwanted yaw (pitch) moment arm is created equal to sin θ (0.126) times this radius. (The effect of the 10 degree dihedral of each RCS engine does not effect the relative forces.) For every unit of impulse (I) required in the pitch (yaw) direction, a correcting impulse of I tan θ (or 0.1272 I) in yaw (pitch) must be made. This correction is not automatically calculated by the G&C, but is sensed by the attitude hold system and commanded when the system drifts out of limits. Further, since the correction itself is in error by angle θ, a second or third iteration may be necessary.

The penalty for these errors in extra propellant consumed will be much greater in the AES mission than it is for Apollo. Mission 1, for example, calls for the following propellant requirements (exclusive of this correction) for housekeeping and experimental needs without early quad failure:

Axial translation:　175 lbm

Roll:　527 lbm

Pitch:　710 lbm

Yaw:　713 lbm

Circumferential location of the clusters does not affect translation and roll. However, the 1423 lbm of pitch and yaw propellant creates the need for the following additional propellant:

first order (0.1272)(1423):   181 lbm

second order (0.1272)(181):   23 lbm

third order (0.1272)(23):   3 lbm
                              ─────────
                              207 lbm

All of the AES service module RCS quad designs analyzed to this point require slightly different designs for the 70 degree sectors (II and V) than for the 60 degree sectors (III and VI). Therefore, the only reason for retaining a 7 degree, 15 min circumferential rotation of the engine cluster is no longer valid. The AES circumferential location of the service module RCS engine clusters will, therefore, be moved to the Y and Z axes to realize the propellant savings indicated in the preceding discussion.

EFFECT OF AES ENVIRONMENTS

AES CM and SM reaction control subsystems will be subjected to space environment and propellants for durations approximately three times longer than the Apollo missions. Because of this, it was necessary to investigate the subsystems' capabilities to withstand these increased exposures. A study of the subsystems was conducted in which each Apollo Block II component was examined to identify the materials used, the function of each material, and the anticipated exposure to propellants, vacuum, temperature, radiation, and meteoroids. A literature search was made to obtain the latest published information on metallic and nonmetallic material compatibility limitations. This information was correlated to the extent that data were available. The actual space environment and the effect of meteoroids is reported in SID 65-1534.

The more important factors characterizing the space environment are vacuum, penetrating and ultraviolet radiation, micrometeorites (see report SID 65-1534), and thermal cycling. Information on the effects of these environments is generally available on the basis of an individual parameter and, to a lesser degree, as a combination of parameters. Data obtained from combined environments is usually of more value, since a combination of parameters results in effects on materials which cannot be extrapolated with single parameters.

VACUUM EFFECTS

The range of gas pressures the spacecraft encounters during AES missions varies from approximately 10^3 torr at the earth surface to less than 10^{-12} torr. The vacuum of space is not necessarily the vacuum to which materials in the spacecraft are exposed, since the specific vacuum environment of a material or component depends on the proximity of other materials and the outgassing of these materials. It is, therefore, somewhat difficult to define the exact level of vacuum to which a material is exposed. For the purposes of this study it was assumed that external or vented surface would be exposed to space vacuum for 45 days.

No problems are anticipated as the result of 45 days' exposure of CSM RCS metallic materials to space vacuum except as noted below:

1. The possibility of cold welding occurring in valve seats where metal-to-metal contact exists undisturbed for long periods of time because of the normal, non-operating function of the valves.

2. The molybdenum disilicide coating on the inside of the SM RCS engine combustion chamber may be degraded through sublimation of the silicon while the chamber is still hot following longer firing times.

RADIATION

Exposure of the AES CSM RCS materials to radiation can be simplified into two factors: penetrating (particle) radiation and ultraviolet radiation. The effects of ultraviolet radiation need be considered only for the CM RCS engine ablative and ceramic materials and for the SM RCS engine combustion chamber and nozzle materials that are exposed to direct or reflected sunlight. These exposed materials (phenolic fiberglass, JTA graphite, molybdenum disilicide, L-605 stainless steel) are all highly resistant to degradation from exposure to ultraviolet radiation.

Penetrating radiation, from whatever source, affects a material only to the extent to which it is absorbed. Therefore, the effects should be evaluated on the basis of absorbed dosage. It is estimated that the maximum dosage for the RCS will be 1.2×10^4 rads for the worst case AES mission. The external-surface dosage for the spacecraft will be established when exposure to solar winds, etc., is established. However, because surface-dosage effects are limited to 1 to 3 mils in depth from the exposed surface, no absorption of penetrating radiation is anticipated.

All metals used in the CSM RCS are relatively resistant to radiation (degradation threshold is greater than 10^{12} rads), and no problems are anticipated for AES exposures.

Of the nonmetallics used in the CSM RCS, Teflon TFE has the lowest threshold of degradation when exposed to radiation. Teflon is used as a seal or insulating material in the majority of the CSM RCS components. The most critical use is as a bladder/expulsion device in the propellant tanks, where the radiation dosage could be as high as 3.75×10^3 rads. The bladder is laminated TFE and FEP Teflon. The threshold of degradation is reported to be approximately 7×10^5 rads in a 10^{-6} torr vacuum and 5×10^4 rads at 760 torr (air). No information was available as to what the threshold of degradation might be in the presence of a strong oxidizer (nitrogen tetroxide).

TEMPERATURE CYCLING

The longer AES missions will subject all CSM RCS components to increased thermal cycling. Thermal cycling results in mechanical flexing of interfacing materials having different coefficients of thermal expansion and internal flexing of nonhomogeneous materials such as ablative laminates. Temperature variations also induce pressure cycling in locked up fluid systems.

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Of particular concern are the components of the CM RCS that are non-operative until the initiation of the reentry phase. While the fatigue characteristics of most materials are well established, the influence factors of application and function dictate the examination of all materials in each component for potential problems. One critical component is the burst diaphragm valve that isolates CM RCS propellants from the downstream system until the beginning of reentry.

The ablative material used in the CM RCS engine and nozzle extension is a phenolic fiberglass laminate and can have undetected minor delaminations which are sealed off during reimpregnation or epoxy coating processes. From Apollo qualification testing results, it appears that temperature cycling causes existing delamination to propagate or creates new delaminations, which permit gas leakage through the combustion chamber walls. The capability of phenolic fiberglass laminate to withstand the anticipated temperature cycling of AES missions requires further evaluation.

PROPELLANT EXPOSURE

The range of exposure of RCS materials to propellant can vary from liquid propellant exposure for the full mission duration to short-time propellant vapor exposure such as might occur to external surfaces of an engine valve during acceptance testing. The difference between the effect of propellant vapors on materials and the effect of liquids is not well documented; however, Apollo components are being tested in both according to anticipated exposures. Increased exposure times for AES missions would require added testing for these components.

The propellants used in the CM RCS are monomethylhydrazine (MMH) and nitrogen tetroxide (NTO). The SM RCS uses a 50-50 unsymmetrical dimethylhydrazine and hydrazine mixture (Aerozine 50) as the fuel and NTO as the oxidizer. In general, all metals and nonmetals that are compatible with NTO are also compatible with these two fuels. One notable exception is the material Resistozine 88, which is not compatible with either MMH or Aerozine 50. The metals used in CSM RCS components have been selected for their resistance to propellants; however, reliable propellant exposure limits data were not found for many of the metals used in these components.

Propellants can dissolve, attack, and decompose nonmetals, causing severe degradation of their physical properties, or they can completely destroy the materials. Wherever possible, therefore, nonmetals in the component should not be excessively exposed to propellant. For example, where compression-set and volume-change limitations exist in gasket and seal applications, nonmetals can be enclosed between two metal surfaces

with only a small portion exposed to the propellant. The various categories of Teflon are among the nonmetals that are most resistant to and most stable in propellants; however, NTO does permeate this type of material and is also absorbed by it. Also, resistance of Teflon to radiation exposure is lower in air than it is in vacuum, which leads to the suspicion that resistance would be low in a strong oxidizing atmosphere (NTO).

COMBINED ENVIRONMENTS

The largest area of unknowns in the CSM RCS material compatibility study is the effect of combined environments. The literature reports experience with individual environment parameters on a large number of materials and, to some degree, the combination of two parameters such as vacuum and temperature or vacuum and radiation. Experience with the combination of vacuum, temperature, and radiation is available for some materials. Information concerning the effect of combined exposure is inadequate or not available for most of the materials used in the CSM RCS.

Of primary concern are those applications where exposure of a material nears the limit for an individual environment. For instance, elastomers containing plasticizers become brittle in vacuum. High temperatures increase the rate of embrittlement, and radiation will often accelerate that rate. As noted previously, Teflon is a relatively stable plastic in the individual environments of temperature, vacuum, radiation, or propellants and has been tested quite extensively for each; however, the effect of radiation on Teflon when exposed to hot nitrogen tetroxide has not been established.

Also, as previously noted, phenolic fiberglass laminate as used in the CM RCS engine may be sensitive to long periods of thermal cycling. Apollo will examine the effect of thermal cycling in vacuum for a period of 14 days during qualification testing but will not determine the combined effect of thermal cycling, vacuum, and radiation such as is anticipated during the 45-day AES missions.

AES CSM-RCS CONFIGURATION SUMMARY

The recommended configuration for the AES CSM RCS is shown in Figure 62 for service module sectors III and VI. The configuration for sectors II and V is identical, except that it is designed to accommodate a 70-degree, rather than a 60-degree SM sector. Also, the engine cluster is on the other side of the quad door and closer to the sector centerline; the engine quads are relocated to the Y and Z axes.

Four LEM oxidizer tanks are used in each quad, two on one side carrying oxidizer, and two on the other carrying fuel. The propellant outlet flange design of these tanks has been modified so that one tank design can be used in all locations without making plumbing difficult. The preliminary concept for this change is shown in Spacecraft Design Summary (report SID 65-1529).

Each tank outlet contains provisions for a fill line, a vent line, and a feed line. The feed line is routed to a propellant solenoid valve before being manifolded to the feed line from its mating tank. The two fill and two vent lines from mating propellant tanks are led directly to a servicing panel on the service module skin which is underneath the lines; this panel also contains: (1) a servicing line from the regulated helium supply which pressurizes the propellant tanks, (2) the high-pressure helium fill that services the two helium tanks on that side of the pressurization system, and (3) a GSE test point from the propellant manifold downstream of the propellant solenoid valves. Just downstream of the propellant manifold, a filter is located to prevent contaminating particles from reaching the engines. The present location of the filter on the fuel side is tentative; further investigation may show that it is desirable to move it further downstream closer to the engine cluster.

The pressurization system is schematically the same as system P-3, described previously. The helium tanks are mounted in pairs above the upper dome of the SPS tanks, and the pressurization components are mounted on the two face plates which are part of the helium tank supports. Test points protrude on small stubs of tubing and are not accessible after the quad is installed in the SM. Further investigation is needed to determine if any of these test points must be accessible after SM assembly, and to provide for them if necessary. The emergency helium explosive valve between the two halves of the pressurization system is presently mounted on

a small loop of tubing that locates the valve near a door in the SM skin to permit installation of the Apollo common initiator late in the launch count-down.

Block II SM RCS engine design will be satisfactory for AES with minor modifications, depending on the outcome of the Block II qualification program and the results of future AES studies.

Block II CM RCS system design will be used for AES with minor modifications in selected areas to improve life-extension capabilities of the system.

HELIUM SPHERES (X_S 348.75)

FUEL

FILTER

PROPELLANT M

X_S 338.50

C   C

SPS SUPPORT FITTING
(ON UPPER BULKHEAD -REF.)

X_S 320.970

(SAME AS APOLLO

FILTER

OXIDIZER

5.50
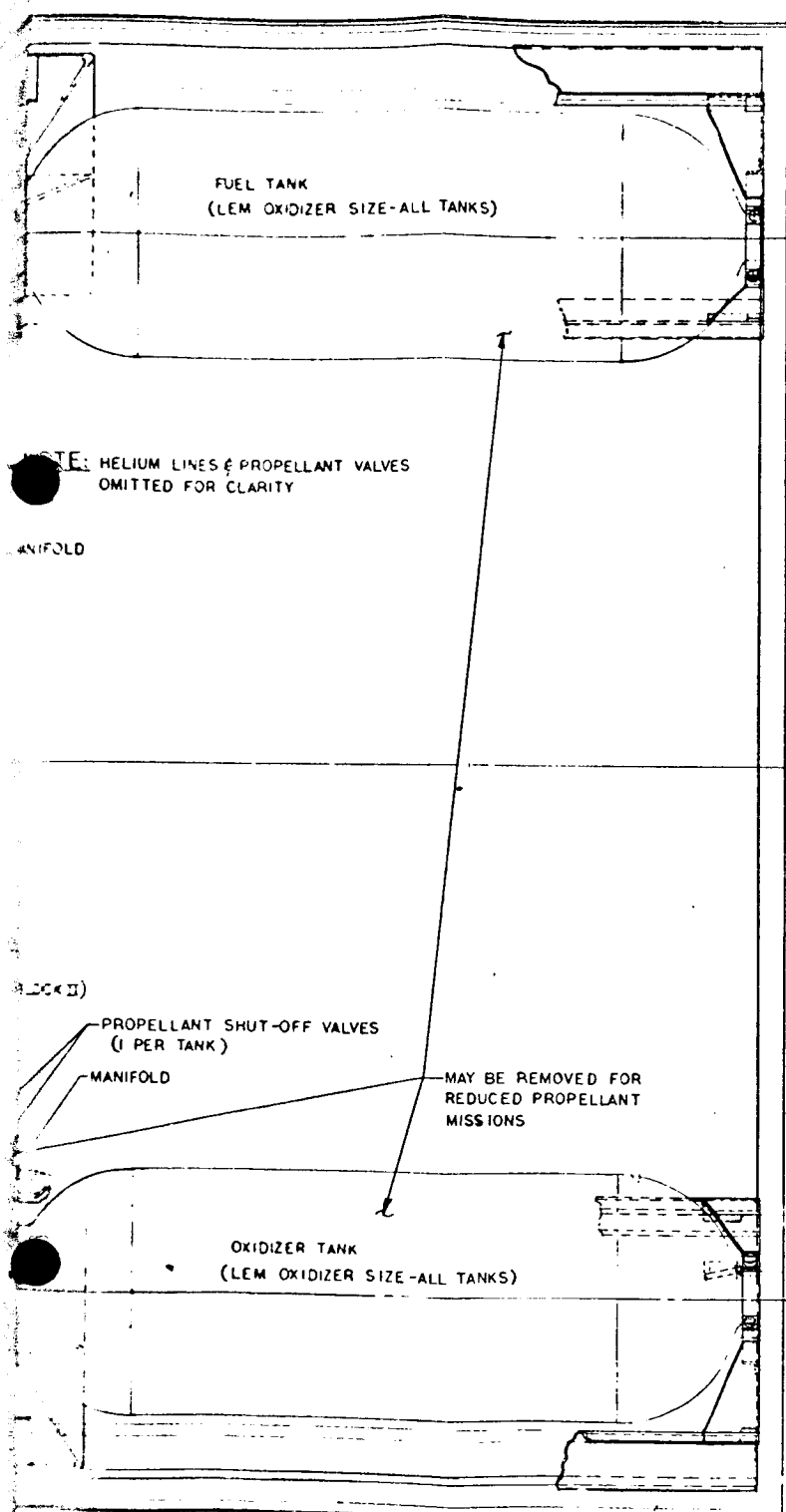
X_S 355.00 X_S 314.60

FUEL FILL OR VENT
HELIUM FILL (HIGH & LOW PRESS.)
FUEL VENT OR FILL
VALVE TEST PT.

FUEL MANIFOLD
& FILTER (IN-LINE)

FUEL

HELIUM TANK
SUPPORT SOCKET
(4 PLCS)

HEL
CRO
LIN

FUEL TANK
(LEM OXIDIZER SIZE-ALL TANKS)

NOTE: HELIUM LINES & PROPELLANT VALVES
OMITTED FOR CLARITY

MANIFOLD

C̶ PANEL

SPS

TP I

MAY BE REMOVED FOR
REDUCED PROPELLANT
MISSIONS

(BLOCK II)

PROPELLANT SHUT-OFF VALVES
(I PER TANK)

MANIFOLD

ROCKET MOTOR HOUSING

MAY BE REMOVED FOR
REDUCED PROPELLANT
MISSIONS

+Z

OXIDIZER TANK
(LEM OXIDIZER SIZE-ALL TANKS)

OXIDIZER

VALVE TEST PT.
OXID. VENT OR FILL
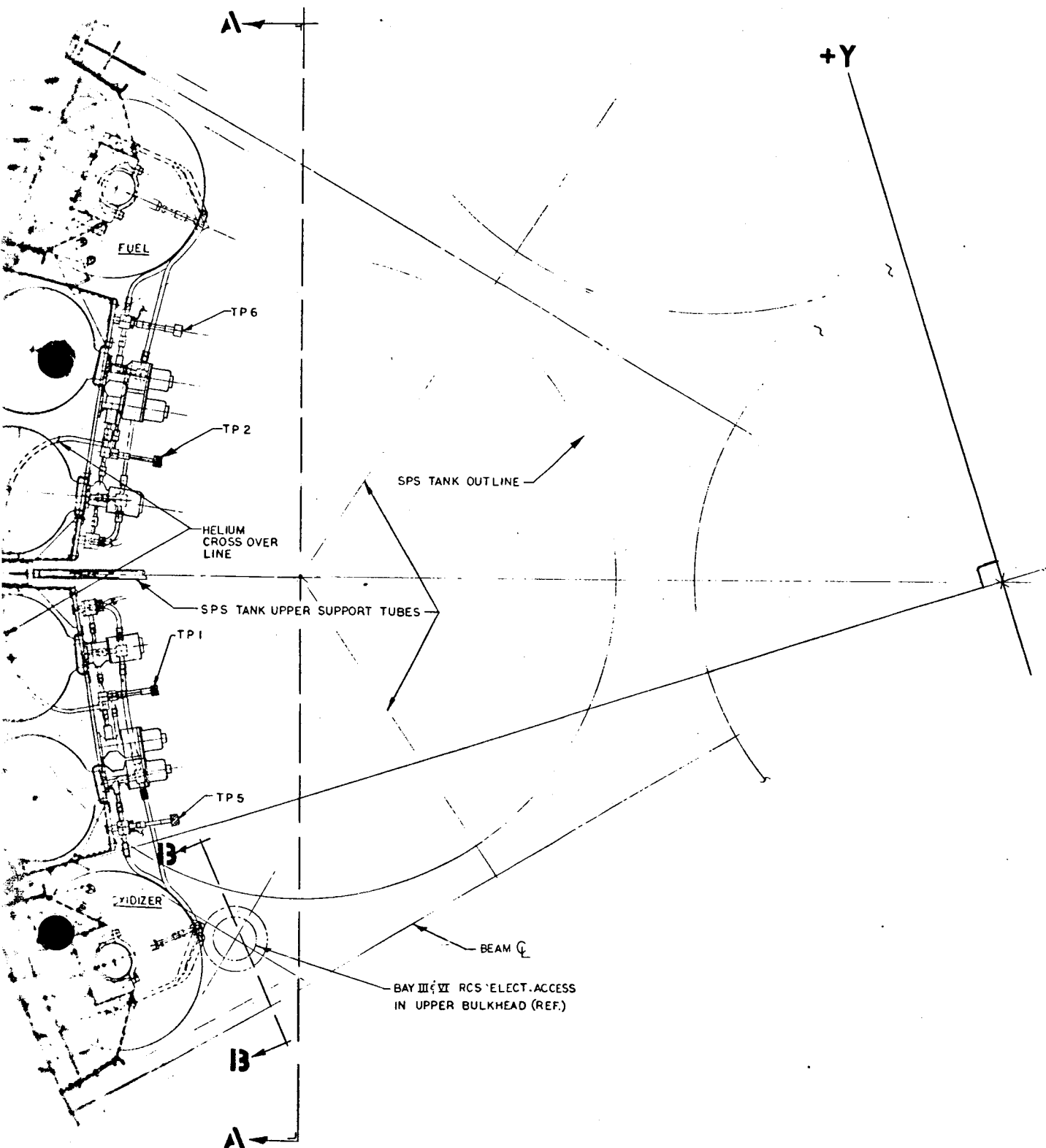HELIUM FILL (HIGH & LOW PRESS.)
OXID. FILL OR VENT

VIEW A A

X_s 269.300
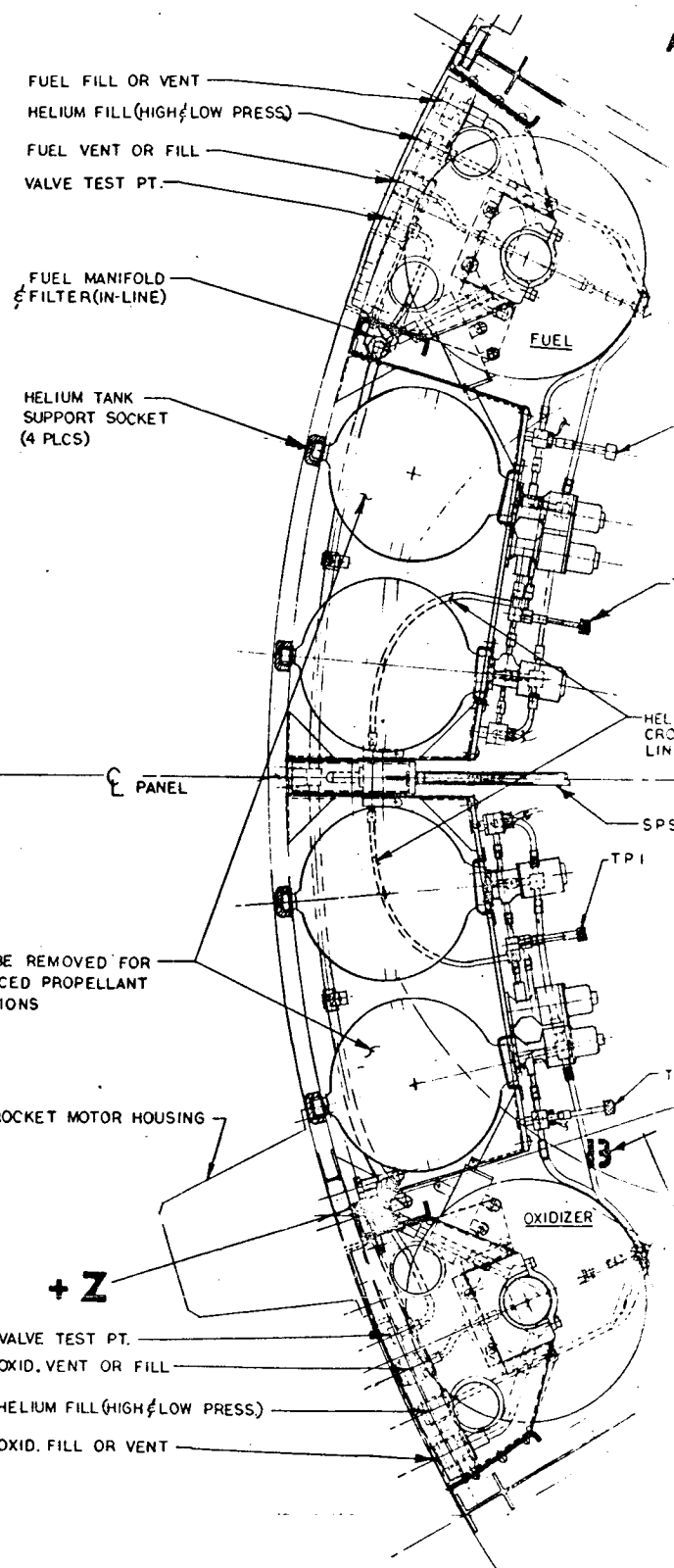
Figure 62. RCS Panel Layout - Bay III and VI of SM

AES SM PROPELLANT GAUGING SYSTEMS

Because of the critical control capability supplied to the vehicle by the service module reaction control system, it is extremely important that the quantity of propellant in the system be known at all times. In order to provide astronaut visibility of the quantity and ratio of oxidizer-to-fuel available to the service module RCS quads during a mission, a system capable of gauging the quantity of propellant in each tank under zero gravity and for extended periods of time is required. Such a propellant quantity gauging system (PQGS) must be adaptable to multiple tanks in the service module RCS quad, must be physically capable of mounting on the tanks or in close proximity, must be compatible with the propellant tank geometry, and must provide highly accurate readings. For AES use, direction was received that the gauging system shall be nonnuclear, in order to avoid interference with experiments.

From the results of the studies, it is apparent that there are several gauging concepts which are of interest to AES. A summary of the more important system concepts proposed by the various companies is given in Table 61. It can be seen from this table that all the proposed systems except that of General Nucleonics require some modification to the propellant tanks. Also, the General Nucleonics system has less potential error sources than the other concepts. The TRW system does have some gamma radiation exterior to the propellant tank and therefore may offer some potential problem with AES experiments. The Bendix and Honeywell systems are presently conceptual only and require considerably more development than any of the others. The Acoustica and Simmonds concepts are similar. The two concepts which appear worthy of further interest for AES are the x-ray direct mass measurement technique as expressed by General Nucleonics, and the pressure-volume acoustic technique as expressed by both Acoustica or Simmonds.

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Table 1.1. ... State of the Art Investigation Summary

GAUGING SYSTEM	Gauging Error	Electric Power (Watts)	System Weight (Lbs)	Tank Modification	Status	Radiation	Comments
Acoustica	±0.5%	48	46	Tank access flange	Air Force test complete by 9/1/66.	None	Reference vol. gas and ullage gas could be different result-ing in system error.
General Nucleonics	±1.5%	32	24	None	Unknown	Yes Krypton-85 gamma	Similar to Giannini
Giannini (present)	±2.0%	55	35	None	Qual. complete by 4/1/66.	Yes Cobalt-60 gamma	Most advanced stage of development, does not require tank modification.
Giannini (improved)	±1.5%	32	24	None	Laboratory testing	Yes Cobalt-60 gamma	Expected to be more reliable than pre-sent system.
Simmonds (mole-metric only)	±0.5%	12	28	Pressure fittings	Unknown	None	Concept only, no known development plan.
TRW/STL	±0.5%	18	25	Pressure fittings	In develop-ment for AEC	None exterior to tank, Kryp-ton-85 Beta in ullage	Full scale hardware in test. Plan to use actual propel-lants. Possible pro-blems with Krypton, permeation is being investigated.

① Based on Block II Apollo - not extrapolated to AES.

NOTE: Information herein is stated by manufacturer and not verified by NAA.

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CONCLUSIONS

During the study, effort was expended to evaluate and define the requirements for the command module and service module reaction control subsystems. The primary ground rule for this effort was that the Apollo Block II is the initial baseline configuration and that recommended changes to the reaction control subsystems should only be the result of AES house-keeping mission considerations. On this major premise, the Block II subsystem designs were carefully scrutinized, AES mission environments and requirements estimated, and the capabilities of the Block II designs compared to the requirements. By this procedure, necessary design changes to the subsystems were identified.

Work conducted on the command and service module reaction control subsystems generally encompassed four areas:

1. The definition of Block II subsystems status and the comparison of subsystem capabilities with AES mission environments

2. The definition of AES propellant requirements for the service module RCS, an evaluation of the service module RCS-engine requirements, and the definition of a service module RCS configuration for AES

3. The evaluation of propellant gaging and test requirements for the subsystems

4. The support to areas interfacing with the reaction control subsystems

The design of the Block II reaction control subsystems and components was reviewed in detail, and the current development or qualification status of each component was assessed. The Block II subsystems have not and will not be evaluated for propellant and space exposure times corresponding to AES mission requirements under the Apollo program. Although some components appear marginal for AES use, a materials compatibility study found no confirmed instance where a material used in the Apollo Block II CSM RCS components would be unacceptable for the longer AES space and propellant exposure. The study did identify a number of materials for which exposure limits were not adequately defined. For these materials, either the test exposures were less than 45 days, the effect of thermal cycling was not

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determined, or the effects of combined exposures were not explored. Material compatibility tests also had not been conducted in some instances. The area representing the least confidence in meeting AES requirements is the effects of combined exposures, such as: thermal cycling in vacuum, thermal/pressure cycling in propellants, and radiation in the presence of a strong oxidizer. Testing is required in these areas.

Service module RCS propellant requirements were calculated for the four AES reference missions. Reference mission 1 (the most demanding) required 1603 pounds-mass propellant for housekeeping and 422 pounds-mass for experiments for a total of 2025 pounds-mass. The propellant available from Apollo and LEM RCS fuel and oxidizer-size tanks, after deduction of losses, was calculated, and trade-off studies of various combinations of tank sizes were performed. Use of two LEM oxidizer propellant tanks for fuel and two for oxidizer permits loading of 2336 pounds-mass of available propellant at the oxidizer-fuel ratio predicted for Reference mission 1. Arrangements of propellant tanks within the quad were also traded off; placement of two tanks on each side of a sector-wide door that extends down to the present ECS radiator is the recommended design.

Trade-offs in reliability, complexity, and weight were made between five candidate service module RCS pressurization systems. One system (P-3) appeared attractive. The fuel and oxidizer tanks were pressurized independently, except for an emergency interconnect between the check valve midpoints. In this design, if one side of the helium system fails at liftoff, the propellant in all four tanks could be expelled by the two helium tanks on the other side, but the last propellant would have a final system pressure of only 136 psia.

The Apollo Block II service module RCS engine cluster design seems suitable for AES except, perhaps, for heater capacity. The cluster location for AES, however, has been rotated to the Y and Z axes, eliminating the present Block II 7°15' offset; this results in a significant saving in propellant requirements.

Basic service module RCS propellant quantity gaging system requirements for AES were established. To accomplish this, a survey of the state of the art of gaging systems was carried out. The following gaging system concepts were evaluated: pressure-volume acoustic, trace gas, x-ray, static pressure, and electrical resonance. The x-ray and pressure-volume concepts for nonnuclear gaging appeared to be the most attractive for AES; they will, however, require development, verification, and qualification testing.